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BIMODAL SATELLITE MISSION STUDY

Capt Fred Kennedy
Dr Vlad Vanek

September 1993

Final Report

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14. Abstract While space system launch and operational costs continue to rise, funding levels for research and procurement are declining drastically. Placing a satellite in geosynchronous earth orbit (GEO) can cost as much as $77,000/kg of payload. Military systems such as MILSTAR, a jam-resistant secure communications satellite, are being examined for redesign in order to place them on smaller, less expensive launch vehicles. Such a stepdown (e.g. from a Titan IV to an Atlas booster) decreases payload capability but lowers launch costs. The decrease in cost can be significant: MILSTAR'S baseline launcher, Titan IV/Centaur, is estimate to cost as much as $400M. Moving the payload onto a Titan III or Atlas IIAS could save $250M or more.

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Bimodal Satellite Mission Study

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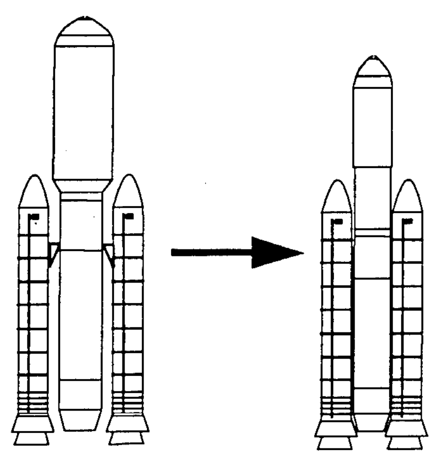
2 September 1993

Executive Summary

While space system launch and operational costs continue to rise, funding levels for research and procurement are declining drastically. Placing a satellite in geosynchronous earth orbit (GEO) can cost as much as $77,000/kg of payload. Military systems such as MILSTAR, a jam-resistant secure communications satellite, are being examined for redesign in order to place them on smaller, less expensive launch vehicles. Such a stepdown (e.g. from a Titan IV to an Atlas booster) decreases payload capability but lowers launch costs. The decrease in cost can be significant: MILSTAR's baseline launcher, Titan IV/Centaur, is estimated to cost as much as $400M. Moving the payload onto a Titan III or Atlas IIAS could save $250M or more (Fig. 1).

Launch Vehicle Stepdown



Titan IV/Centaur
$400M/launch

Titan III/NUS
$145M/launch

Figure 1 Cost figures were taken from the *Launch Vehicles Summary for JPL Mission Planning*, JPL D-6936 Rev. C, February 1993.

Achieving launch vehicle stepdown without degrading a satellite's capability demands either lighter subsystem components or higher-performance propulsion. The authors

propose the use of a bimodal power and propulsion system that would provide (1) orbital transfer and on-orbit maneuver capability to a satellite as well as (2) electrical power for housekeeping and instrumentation. Current satellite platforms use separate propulsion and power subsystems; stationkeeping and maneuvering are usually performed by monopropellant or bipropellant hydrazine thrusters. Power is commonly generated via solar photovoltaic arrays and stored in batteries. A bimodal system would be based on either a nuclear reactor or a solar array coupled to a thermal storage bed. Either would provide heat for thermal propulsion and thermal-to-electric power conversion (Fig. 2).

Bimodal Satellite Concept

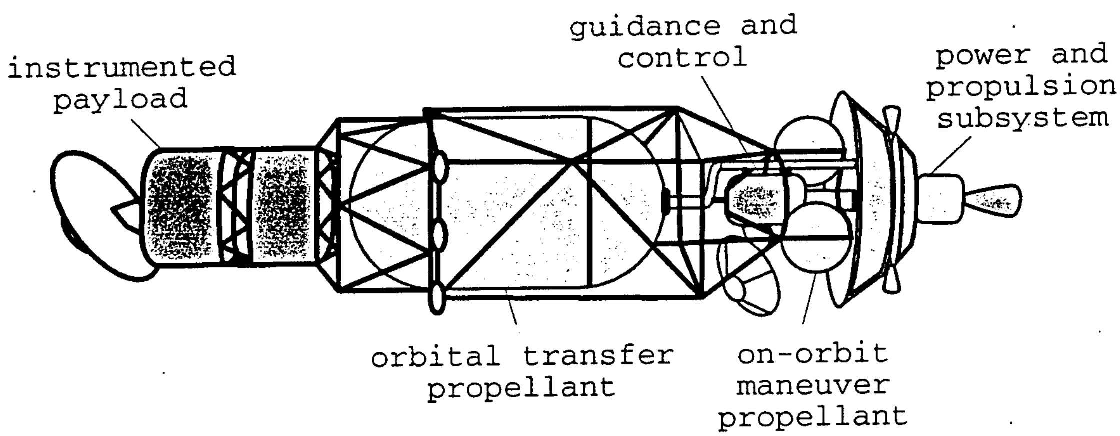


Figure 2 A bimodal power and propulsion unit would functionally replace upper stages (e.g. Centaur, TOS, IUS), the onboard solar photovoltaics and batteries, maneuvering and attitude control thrusters, besides a portion of the guidance and thermal management subsystems.

It is the objective of this study to examine a number of current and future missions--with both civilian and military applications--to determine the requirements that a bimodal system would have to meet to be effective. A prioritized list of these requirements includes *reduction in launch and operational costs, enhancement of current missions*, and *enabling of future missions*.

The key results of the study are:

(1) **A number of launch vehicle stepdowns are enabled by a bimodal power and propulsion system operating at specific impulses of between 600 and 940 s.** Geosynchronous orbit payloads normally requiring Titan IV/Centaur (*T4/C*) can be moved down to a Titan III/NUS (No Upper Stage) if the bimodal propulsion system is

permitted to fire at altitudes of 475 km or less. Stepdown from *T4/C* to Atlas IIAS is made possible only by using high thrust-to-weight, high-I_{sp} (940 s) systems at startup altitudes under 250 km. A stepdown from Atlas IIAS to the smaller Delta II 7920 rocket is achievable by all systems examined at startup altitudes as high as 1000 km.

(2) **Bimodal power and propulsion enables extremely efficient on-orbit maneuvering.** GEO repositioning and sun-synchronous orbit repositioning are two cases examined in this report. The delta-V requirements for GEO repositioning are small (<200 m/s for 30°/day longitude shifts). A bimodal system operating at I_{sp}'s of 600 to 940 s cut total propellant use by a *factor of three*; electric propulsion systems would provide even higher specific impulse and reduce propellant mass needs to less than a sixth of that required by conventional bipropellant hydrazine thrusters. These reductions also apply in the case of sun-synchronous orbit repositioning, where high delta-V requirements (2000 m/s in some cases) preclude current satellite users from performing these maneuvers.

(3) **Bimodal reactor systems must be designed that are smaller than current power systems.** The work performed in section 3.0 was independent of bimodal system mass and leaves open the question of whether the system itself would be too heavy to provide benefit; section 4.0 was devoted to determining the capabilities of bimodal systems having masses ranging between 600 and 1600 kg. There was significant agreement between sections 3.0 and 4.0: Titan IV/Centaur payloads could be stepped down to either Titan III/NUS or Atlas IIAS, although the latter would require a high-I_{sp} system (940 s) in addition to a very lightweight reactor system (600 kg or less).

(4) **The choice of initial orbit can dramatically affect bimodal payload deliveries to the destination orbit.** Launchers such as Titan III and IV have large throwweights to low orbit. At 185 km, these systems respectively place 14.5 and 22.5 metric tons of payload; unfortunately, these systems cannot currently be restarted and this results in a precipitous decline in their payload delivery capability to higher orbits. At an altitude of 1000 km, Titan III is incapable of placing any mass, while Titan IV can only achieve about 3 metric tons--this is below the ability of Atlas IIAS and Delta. Restricting a bimodal system to operation above some specified minimum altitude could thus seriously handicap it in comparison to systems that are not so restricted.

Recommendations for bimodal system specifications included a reactor system mass of *850 kg* producing *8.5 kW_e* (or an alternative *2000 kg* system with a power production capability of *20 kW_e*), a thermal propulsion system (TPS) thrust level of approximately *80 N (18.2 lbs)*, a TPS specific impulse of *770 s*, and an operating life of *10 years*.

1.0 Introduction

Placing large satellites on-orbit is an expensive proposition. The United States Air Force is currently contemplating two new systems, MILSTAR and FEWS (Follow-On Early Warning System), for deployment in geosynchronous orbit. MILSTAR is a highly survivable, jam-resistant communications satellite that weighs over 4700 kg and requires the largest expendable booster in the US inventory to get it to its operational orbit. That booster, Martin-Marietta's Titan IV/Centaur, is estimated to cost over $400M per launch; the Senate Armed Services Committee "noted that as costs are rising, large payloads are dwindling. They cited the possibility of redesigns that would allow MILSTAR and FEWS satellites to be launched on medium-lift vehicles. The dearth of payloads will lead to slowed procurement rates, driving Titan IV unit costs up." [Ref. 1]

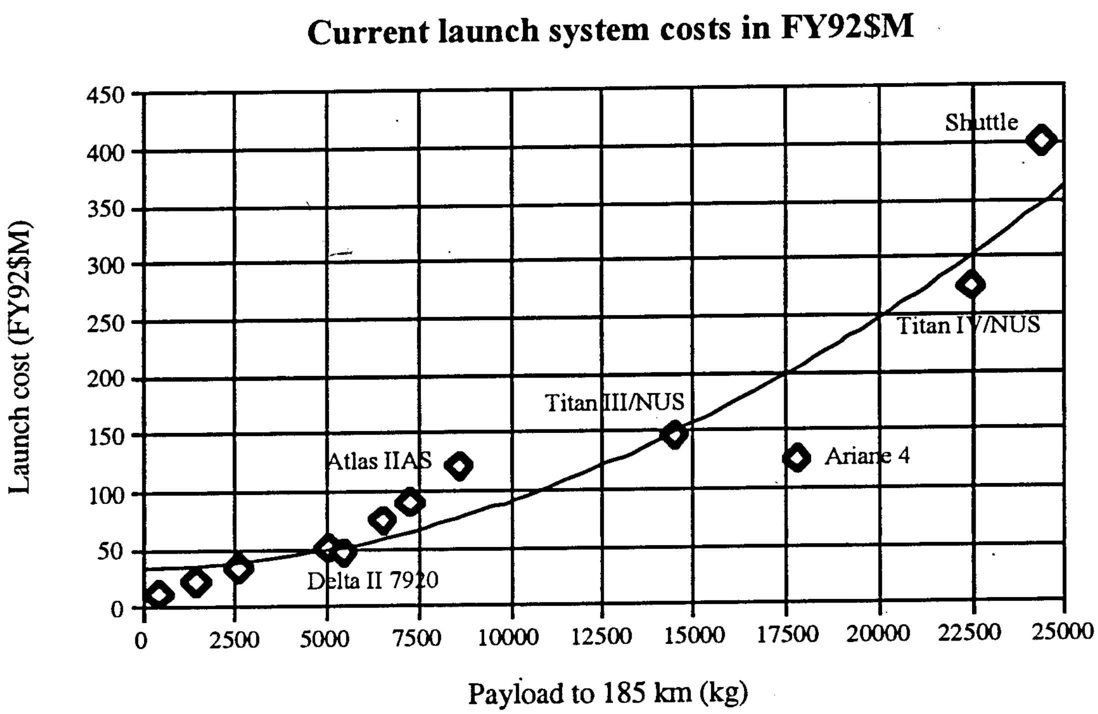


Figure 1.1 The performance of various American boosters is shown versus system cost. The French Ariane 4, which is launched from French Guyana off the South American coast to take advantage of the region's low latitude, is included for comparison. Note that the two data points closest to the origin are Orbital Sciences' Pegasus and Taurus boosters. Cost figures were taken from the *Launch Vehicles Summary for JPL Mission Planning*, JPL D-6936 Rev. C, February 1993.

FEWS is intended to detect and track dim, short-burning targets such as intermediate-range missiles or afterburning aircraft [Ref. 1]. The FEWS satellites are being downsized to allow them to be launched aboard Atlas IIAS boosters instead of the baseline Titan IV/Centaur. As Atlas IIAS can be purchased for $120M, it is conceivable that moving off of Titan will save nearly $300M in launch costs. To achieve this stepdown from a large

booster to a smaller, less expensive one, the FEWS satellite will be deployed in a highly elliptical orbit with an apogee of approximately 11000 km. The onboard hydrazine thrusters will then be used to move the satellite to its operational orbit at GEO. [Ref. 2] It was not possible, however, to step down to Atlas by a change in mission profile alone; a number of instruments originally planned for the platform will be removed in order to meet the more stringent weight requirements.

The MILSTAR and FEWS cases are instructional in that they provide a clue to the apparent direction of future US military satellite design and deployment. While development and launch costs have risen dramatically, the funding available for large projects has declined drastically. Air Force Space Command has proposed Spacelifter, the most recent in a line of heavy booster concepts that are intended to lower launch costs to LEO and other orbits (e.g. the Advanced Launch System and National Launch System). Lower costs are also achievable by improving the power and propulsion systems available to the satellite user. Mass is cost to the space system designer, and there are only several ways to decrease mass requirements. Improvements in subsystem design could lower overall weight (e.g. improved electronics, higher-efficiency photovoltaic arrays, or smaller batteries). Requirements can simply be degraded and lower-priority instruments left off altogether, as in FEWS. For a geosynchronous satellite, an upper stage, such as Centaur, is necessary to move the system from its intermediate to its destination orbit. Often, an apogee kick motor is also required to circularize the orbit once it is reached.

Almost all current transfer stages are based on solid or liquid chemical propulsion, a fact that constrains their performance by limiting the specific impulse, or I_{sp}, of the stage to approximately 450 s [A more detailed explanation of rocket performance can be found in Appendix A]. Specific impulse is essentially a function of exhaust temperature and propellant molecular weight. Chemical systems are energy-limited, deriving their power from combustion. Since a given reaction can liberate only a specified amount of heat--being stoichiometrically determined--there is a ceiling to a chemical system's performance. The Space Shuttle uses hydrogen-oxygen combustion, delivering 455 s of specific impulse, extremely close to the theoretical maximum achievable by this reaction.

Thermal and electric propulsion systems, on the other hand, are power-limited. No chemical reaction takes place. An electric propulsion device converts electrical power to propellant kinetic energy by heating the propellant (through either induction or convection) or by ionizing the exhaust gases and establishing an electromagnetic field to accelerate individual particles. Electric systems are characterized by very high specific impulse (thousands of seconds) but extremely low thrust, typically less than 1 N; thus,

an electric thruster such as an ion engine increases payload but often at the cost of very long transfer times.

A Bimodal Reactor System

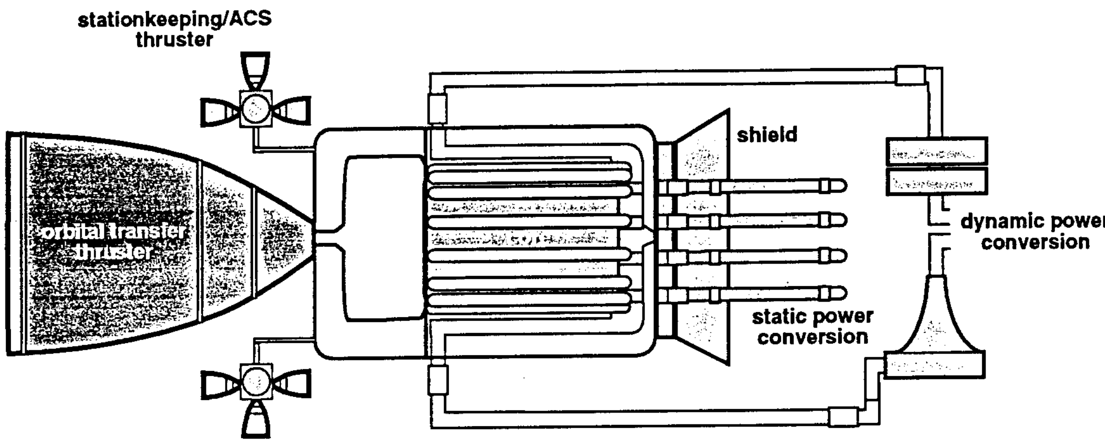


Figure 1.2 A bimodal power and propulsion system consists of a heat source (here, a nuclear reactor), static or dynamic power conversion and distribution systems, a thermal propulsion scheme and optional electric propulsion devices, thermal management, and shielding. This system functionally replaces a number of systems aboard most satellites.

A thermal propulsion system would use a nuclear reactor or a solar concentrator to heat the propellant, normally hydrogen, which has the lowest molecular weight of any element. High thrust is possible, along with specific impulses nearing 1000 s. It is notable that high I_{sp} is gained not because overall exhaust temperature is higher than a comparable chemical system (material limits are the same in both cases), but because the thermal system need not use propellants based on their reaction energy content.

The authors propose a bimodal power and propulsion system, based on a nuclear reactor, to perform orbital transfer and on-orbit maneuvering functions, as well as to provide all onboard power to the satellite (Fig. 1.2). A number of space nuclear power systems have been designed and several have been flown, notably the Radioisotope Thermoelectric Generators (RTG's) placed aboard the Pioneer, Voyager, and Galileo spacecraft. Space nuclear propulsion systems recently gained worldwide attention when the Air Force's Space Nuclear Thermal Propulsion Program was declassified following research during the 1980's; when President Bush announced his Space Exploration Initiative in 1989, intended to return man to the moon and later attempt a voyage to Mars, a number of studies were released that showed the usefulness of nuclear propulsion for these missions.

A bimodal power and propulsion system would use the reactor as an energy source to perform both of its functions. It would (1) heat propellant passed through the core, which would be expanded in a nozzle and expended to provide thrust, and (2) provide a high-temperature heat source for static or dynamic power conversion systems, which would drive onboard instrumentation. It would also be possible to power electric propulsion systems in order to gain a high-efficiency maneuver capability.

By delivering much higher performance than current propulsion systems and by utilizing the same unit to provide power, it will be shown in the following sections that a bimodal system can *reduce launch and operational costs*, *enhance existing missions*, and *enable demanding missions* that could be fielded in the near future.

2.0 Objectives, Requirements, Constraints, and Mission Elements

Figure 2.1 illustrates the interdependency of mission objectives, requirements, and constraints, and how the various mission elements fit into the bimodal satellite architecture. Objectives are delineated in section 2.1, requirements and constraints in 2.2, and elements in 2.3.

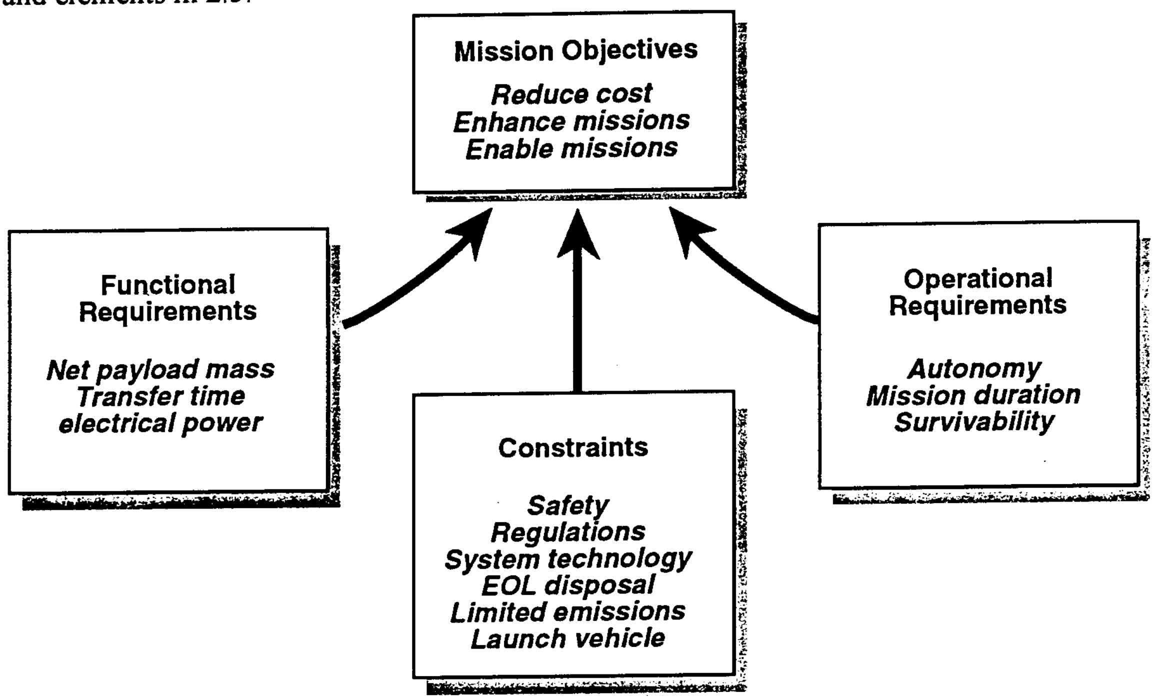


Figure 2.1 The bimodal power and propulsion system will be designed with three major objectives in mind: Reduce launch and operational costs, enhance the capabilities of existing missions, and provide a level of performance that enables new missions.

2.1 Mission Objectives

The objectives of the bimodal satellite concept are to:

1. *Significantly reduce the cost of existing military (and commercial) missions* by:
    - a) lowering launch costs to geosynchronous or high earth orbits
    - b) providing longer life to satellite platforms
    - c) providing increased capabilities to individual satellites
    - d) enabling a standardized satellite bus
2. *Enhance existing military missions* by:
    - a) improving on-orbit maneuverability
    - b) increasing payload delivery
    - c) increasing on-board power

3. *Enable new missions* by:

a) providing greater payload weights to any orbit

b) allowing power levels as high as 100 kW_e

The propulsion capabilities of a bimodal-powered satellite are driven by payload mass and transfer time requirements to orbits of interest. Electrical power capabilities are derived from satellite platform power needs (such as housekeeping and instrument power). This study will focus on several of these objectives and thereby attempt to define requirements for a bimodal power and propulsion system.

Note that some of these (e.g. allowing power levels approaching 100 kW_e, providing longer lifetime to satellites) fall into the province of bimodal system design rather than mission analysis. Increasing a particular satellite's capabilities is extremely mission-specific, and will not be addressed within this report.

2.2 Functional and Operational Requirements, and Constraints

a) *Functional requirements* define quantitatively how the bimodal power and propulsion system must perform to meet its objectives. There are three major requirements singled out; the actual value of each of these will be determined by the individual missions:

Net payload mass at the destination orbit (kg)
Transfer time (hours)
Electrical power (kW_e)

b) *Operational requirements* determine how the bimodal satellite operates and how users will control it. There are three operational areas identified for preliminary requirements:

System autonomy
Satellite operational lifetime
Survivability (total dose)

c) *Constraints* limit schedule, cost and mission implementation techniques. The following constraints on bimodal satellite development and deployment have been identified:

Safety (minimum operational orbits, system design)
Regulations
Technology availability
End-Of-Life (EOL) disposal
Limited particle emissions (interference with other systems)
Launch vehicle

2.3 Mission Elements

The bimodal satellite mission consists of a set of elements which are arranged into a mission architecture. In principle, these elements are subjected to trade studies during the design phase to satisfy requirements at least cost. Limited trades are performed in section 3.1-3.4, primarily involving the bimodal subsystem of the satellite and the orbital element. The bimodal satellite mission elements are:

Bimodal subsystem element - the bimodal subsystem hardware and software (bimodal power/propulsion system, propellant vessel and refrigeration, guidance, navigation and control set, communications, etc.).

Payload element - the customer satellite payload

Launch system element - includes the launch facility, launch vehicle(s) required for the bimodal satellite vehicle, as well as interfaces, payload fairings, and associated ground-support equipment and facilities. The satellites are launched by variety of Earth-to-orbit vehicles (Delta II 7920, Atlas II family, Titan III, Titan IV, Ariane, etc.).

Orbit element - comprised of all bimodal satellite vehicle trajectories, including the initial orbital insertion, all on-orbit maneuvers and disposal orbits.

Communication architecture element - must satisfy command, control, and communications requirements for operation of the bimodal satellite. The primary payload may have a separate communications architecture.

Ground system element - comprised of fixed and/or mobile ground stations connected by communication links. This segment allows command and tracking of the bimodal satellite vehicle during its operation.

Mission operations element - consists of those personnel controlling bimodal satellite operations and their attendant policies and procedures.

3.0 Candidate Bimodal Mission Applications

Among the missions investigated were (1) Orbital transfer from low earth orbit to geosynchronous orbit, (2) repositioning satellites in sun-synchronous orbits, (3) repositioning satellites in geosynchronous orbits, and (4) dual-satellite launches.

3.1 LEO-GEO Transfer

It has been stated that a bimodal power and propulsion system realizes cost savings in three specific ways: (1) through stepping down a satellite from a larger to a smaller, less expensive launch vehicle; (2) through increasing system lifetime, thus allowing fewer satellites and launchers during the life of a particular program; and (3) increasing system performance, which allows more tasks to be performed by a single satellite platform, again decreasing the necessary number of satellites and launchers. This section focuses specifically on the first of these: How a bimodal system allows launch vehicle stepdown, i.e., lofting a Titan IV-class payload aboard a Titan III or Atlas IIAS in order to save money associated with launching the larger system. The following is excerpted from *An Autonomous Transport System (ATS): An Application of Bimodal Nuclear Power and Propulsion* [Ref. 3]:

"Geosynchronous earth orbit (GEO) is an extremely important resource for satellite placement; at an altitude of 35,786 km and near-zero inclination, the orbital period is equivalent to that of earth's sidereal day. The essentially stationary ground track that results from this orbit makes it useful for a variety of missions. Communications satellites make up a large portion of the GEO population, although there exist platforms whose missions include boost-phase strategic and intermediate range missile tracking, detection of nuclear detonations, and weather observation. The cost for placing a satellite in GEO is extremely high. While launching to low earth orbit (185 km) results in figures of $10,000/kg and below, transfer to GEO raises this significantly--by as much as a factor of ten.

"The current scheme for placing a US military payload in GEO involves a booster vehicle such as Atlas or Titan, an upper stage (Inertial Upper Stage, Transfer Orbit Stage, Centaur, and the Payload Assist Module), and perhaps even an apogee kick motor to circularize the orbit once GEO is attained. Commercial payloads equipped with upper stages can be placed in GEO by a number of launchers. The most notable of these is the French *Ariane 4* which, due to its low-latitude launch site, is more efficient than vehicles

launching from either US site and has captured more than half of the launch traffic for commercial communications satellites."

Booster/Upper Stage	Launch Cost (FY92$M)	Injected mass at GEO (kg)
Delta II 7925/PAM-D	50	910
Atlas IIAS	120	1060
Titan III/TOS	188	1360
Titan IV(SRMU)/Centaur	400	5220

Table 3.1 US Booster and Upper Stage Performance

Table 3.1 shows the current US booster/upper stage combinations used to move platforms to GEO. Notice that none of the systems save Titan IV/Centaur can place more than 1400 kg in GEO. All costs are in FY92 dollars [Ref. 4]. Note that the cost of launching the maximum Titan IV payload to GEO is almost $77,000/kg. Delta II provides the lowest specific cost: $55,000/kg.

Table 3.2 lists three geosynchronous satellites that are of interest for this portion of the study, including an advanced mission that could be performed early in the next century. Note that, until recently, MILSTAR required approximately 10.5 kW_e from its solar arrays at beginning of life; this has been reduced to ~ 8.4 kW_e. The Follow-On Early Warning System (FEWS) satellite will be placed in a 185 x 11000 km orbit by an Atlas IIAS; the onboard hydrazine thrusters will then be used to move the satellite from its highly elliptical orbit to geosynchronous. A GEO-based Direct Broadcast High-Definition Television (DB HDTV) system would require as much as 100 kW_e in order to reach 0.5-m receivers in users' residences or portable military HDTV sets on the battlefield; lower power levels could be achieved at the cost of additional satellites. This system could weigh between 10000 and 18000 kg [Ref. 5]. There are currently no launchers in the US inventory which can lift this mass to GEO.

Satellite	Mass (kg)	Power (kWe)	Lifetime (yrs)	Launcher	Launch date	Constellation
MILSTAR	4760	~5	10	Titan IV/Cent	Classified	3 w/spare
FEWS	2000	3-5	10	Atlas IIAS	Post-2000	3 w/spare
DB HDTV	*10000*	*20-100*	*10*	*NA*	*Post-2000*	*1 (100 kW_e)*

Table 3.2 Geosynchronous Satellite Platforms and their Characteristics
(NA: None Available in US)

A bimodal power and propulsion device is conceived that would replace the LEO-GEO transfer stage and then operate as the power production system for the GEO-based satellite platform. In addition, all required on-orbit maneuvering, stationkeeping, and end-of-life disposal would be performed by this same bimodal system.

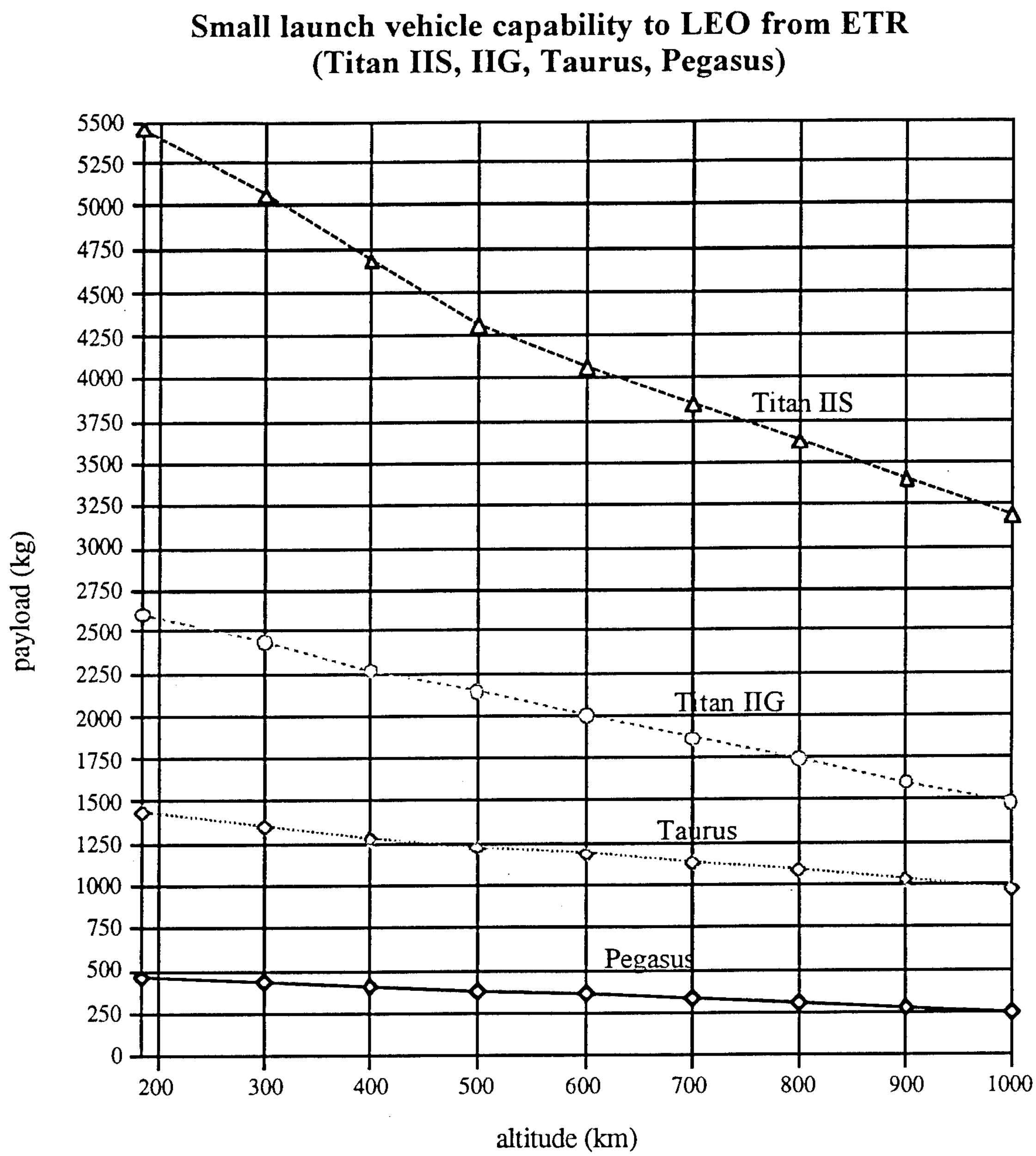


Figure 3.1 Assumptions include launch from Cape Canaveral (ETR) into a 28.5° circular orbit and a Titan II launch capability at ETR (which does not currently exist). Titan IIS is an upgraded Titan configuration that is included for comparison purposes only, since the system is not currently operational.

By fulfilling these various mission requirements with a single integrated system and by allowing launch vehicle stepdown, it should be possible to reduce both launch and operational costs to geosynchronous orbit. In order to deduce requirements for such a system, it was necessary to examine the current launch capabilities of the various boosters in the US inventory. Future heavy booster capabilities were not analyzed.

Intermediate launch vehicle capability to LEO from ETR (Atlas II, IIA, IIAS, Delta II 7920)

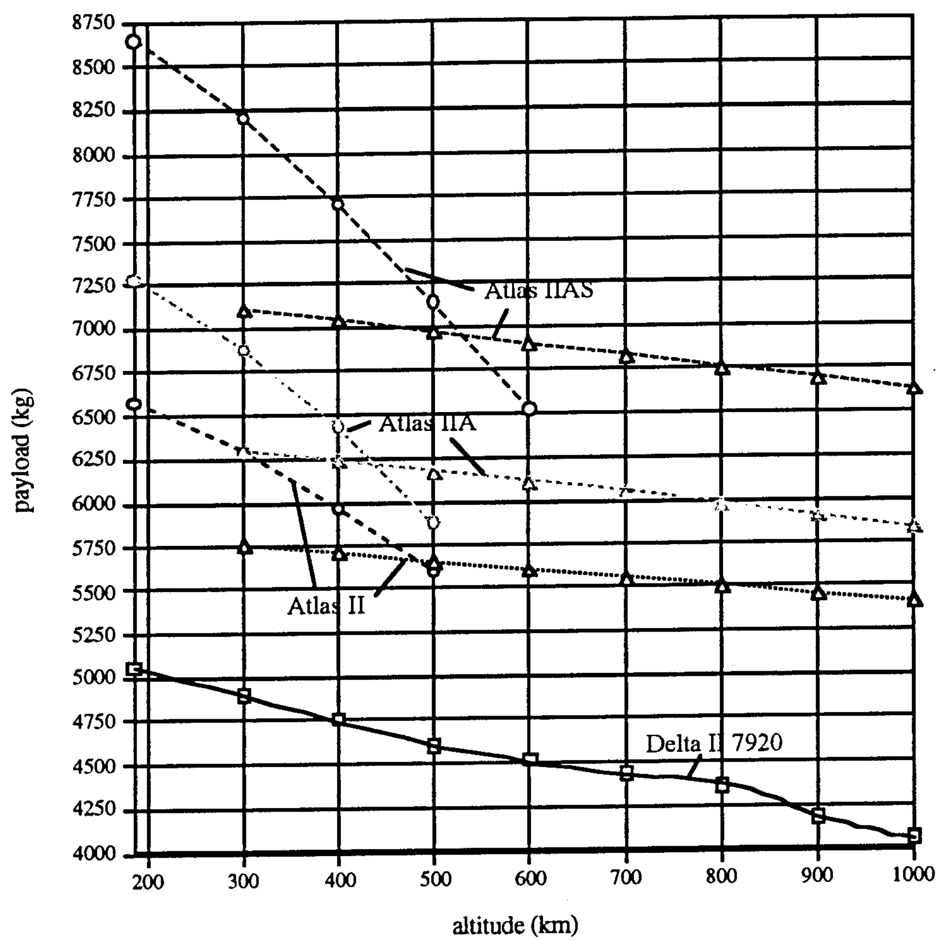


Figure 3.2 Assumptions are the same as in the previous figure (28.5° inclination, circular orbit). The payload capability of these systems degrade fairly gracefully at higher altitudes. There are two sets of performance curves for each booster, corresponding to a single Centaur burn (higher payloads at low altitudes) and a two-burn (curves beginning at 300 km).

The smallest vehicles (Pegasus, Taurus) can lift no more than 1500 kg to 185 km (Fig. 3.1). Given the mass budget analysis performed in Section 4.1, it is clear that these systems will be too small to lift even the bimodal system.

Intermediate vehicles include Delta II and the Atlas family of launchers, placing between 5000 and 8600 kg in 185 km (Fig. 3.2). The Atlas II boosters use a Centaur upper stage

which can be restarted to improve performance at higher altitudes (>500 km). The first set of Atlas curves--up to approximately 500 km--represent the capability of these systems when only one Centaur burn is used; above this altitude, two Centaur burns are needed. Atlas IIAS loses approximately 23% of its performance when the minimum altitude rises from 185 to 1000 km.

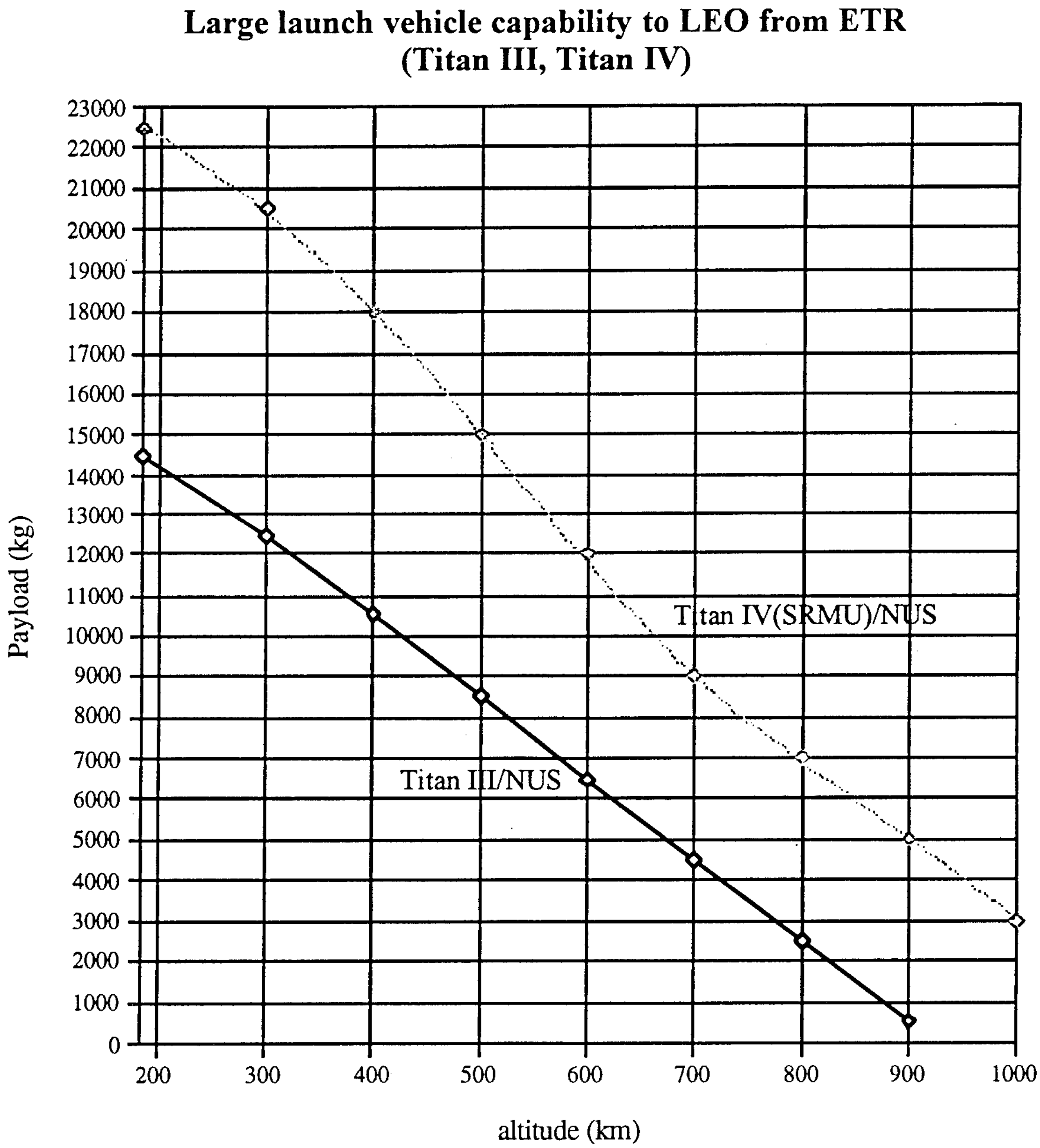


Figure 3.3 Assumptions are the same as in the previous figure (28.5° inclination, circular orbit). Titan IV(SRMU)/NUS denotes a new Titan IV with upgraded solid rocket motors and no upper stage. These systems are the largest expendable vehicles currently available in the United States.

Titan III and Titan IV bound the upper range of US launch capabilities (Fig. 3.3). Their performance declines drastically as altitude increases, falling below the intermediate vehicles above approximately 700 km. Titan IV experiences an 87% decrease in delivered payload over the altitude range of interest; Titan III is incapable of delivering any payload to 1000 km. The Space Shuttle is not included in Figure 3.3 although it does provide up to 25000 kg at 185 km; its performance declines to essentially zero payload at 600 km.

Conventional and Bimodal MILSTAR Missions

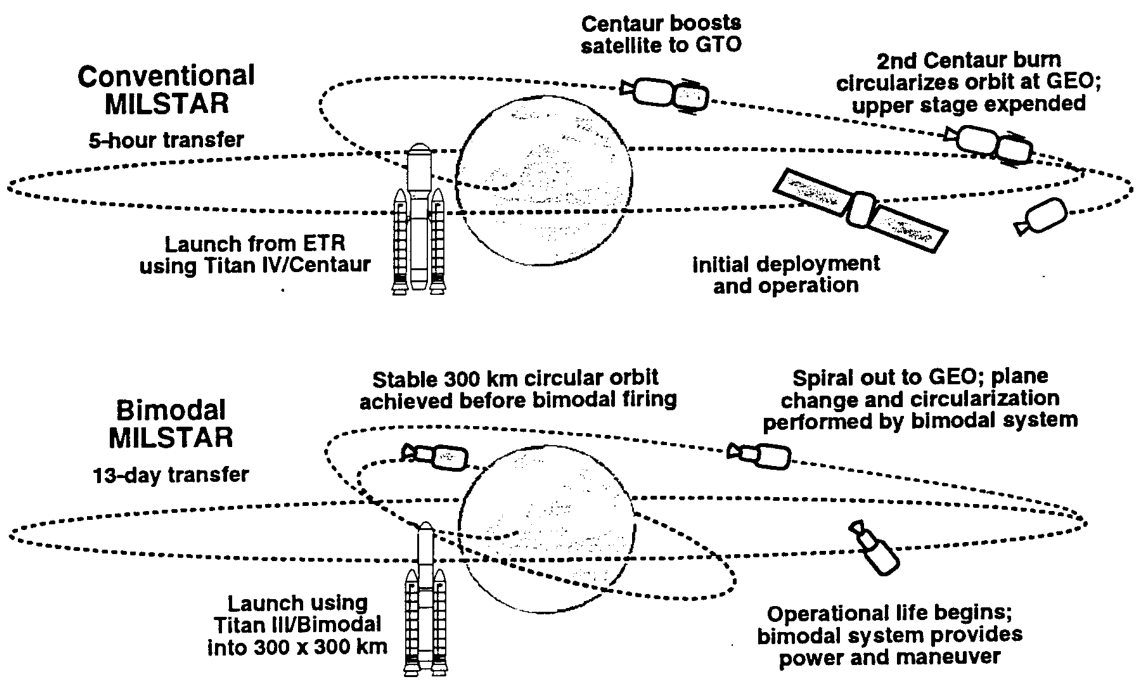


Figure 3.4 The baseline MILSTAR mission requires the launch of a Titan IV/Centaur ($400M/launch) and the Centaur's subsequent boost to geosynchronous transfer, whereupon the orbit is circularized. The bimodal MILSTAR would require only a Titan III launcher ($145M) to boost the system to a low earth orbit. The bimodal propulsion would then transfer the satellite to GEO and begin providing on-orbit power, stationkeeping, and repositioning for MILSTAR. This particular transfer assumes a bimodal Isp of 770 s and a continuous thrust level of 50 N.

The proposed mission profile for a bimodal transfer to GEO is as follows (Fig. 3.4):

(1) Launch bimodal satellite from ETR into a LEO orbit (185-1000 km);

(2) Perform LEO-GEO transfer with a continuous burn using the bimodal power and propulsion system;

(3) Perform circularization and plane change once geosynchronous orbit is attained (this assumption is very conservative, due to the high cost of the plane change--1600 m/s--which could be performed at the same time as the

spiral-out maneuver); and

(4) Begin GEO operations.

A number of parametric studies were performed to determine an optimal bimodal system specific impulse and thrust, given a range of possible transfer time requirements and the desire to stepdown to smaller, less expensive launchers. Electric propulsion systems were not analyzed for this section because their long transfer times fall far outside this study's range of interest (< 4 weeks LEO-GEO). Figures 3.5 and 3.6 illustrate the respective dependence of transfer time and delta-V on startup altitude.

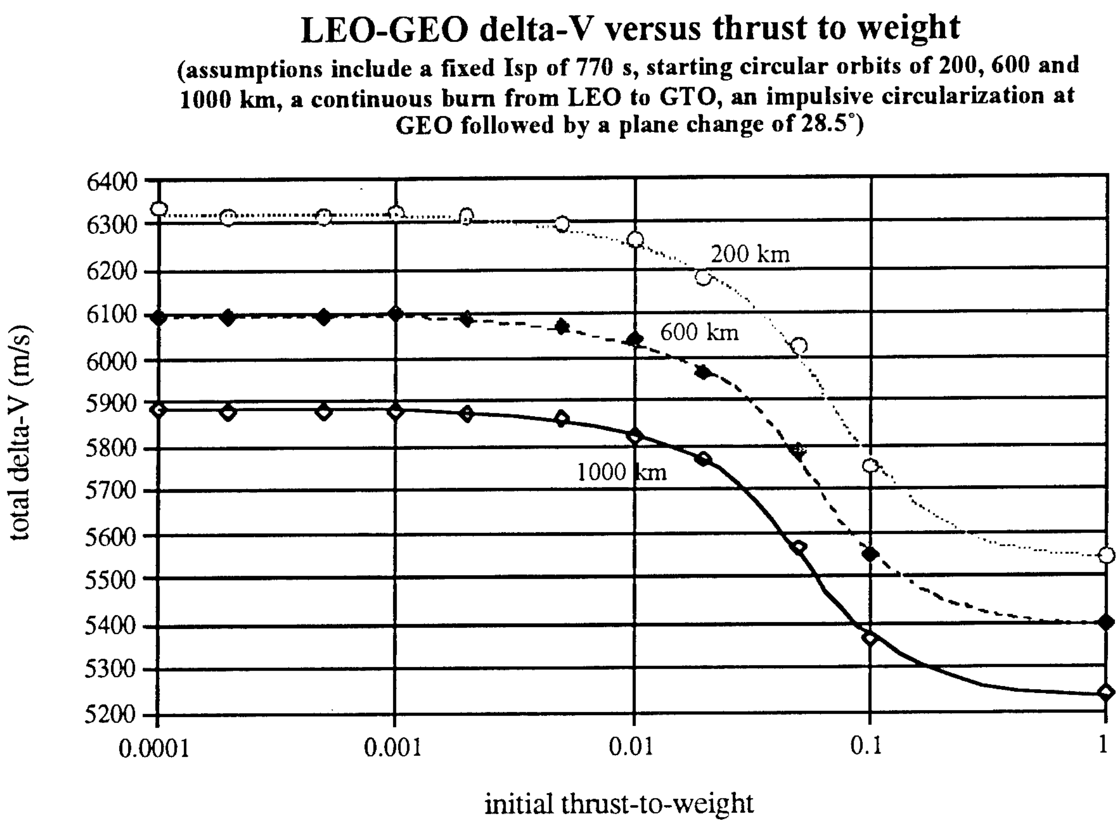


Figure 3.5 There is only a small difference in delta-V requirements among starting altitudes of 200, 600, and 1000 km.

The required LEO-GEO delta-V declines as the startup altitude increases, albeit slowly (Fig. 3.5). There is only a difference of 400 m/s separating a 1000 km start from a 185 km start; this is approximately 6% of the total delta-V requirement. For an injected mass in GEO of 5220 kg (that delivered by a conventional Titan IV(SRMU)/Centaur, increasing the startup altitude from 185 to 1000 km results in an initial mass benefit of ~ 600 kg--that is, moving to the higher altitude requires 600 kg less propellant to place the largest Titan IV payload in GEO. This assumes a propulsion I_{sp} of 770 s. Note that over this same range of altitudes, the total Atlas throwweight to GEO declines by 1960 kg; Titan

IV(SRMU) drops by an even larger amount, 19500 kg. Therefore, the apparent benefit of increasing startup altitude is entirely negated by declining launch vehicle throwweights. This demonstrates the very important fact that startup altitudes must be as low as is considered reasonably safe in order to achieve the larger payloads necessary for launch vehicle stepdown. This point will be further discussed later in this section.

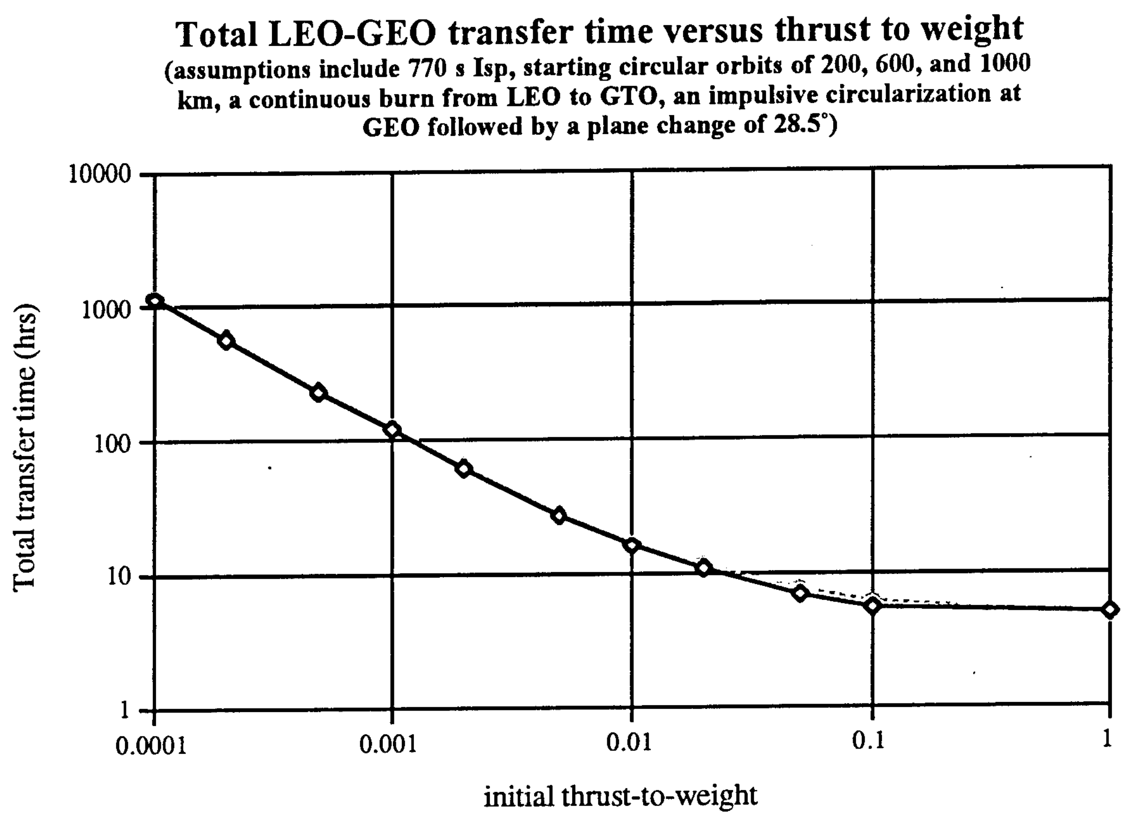


Figure 3.6 The time needed to perform the circularization is low enough, even at T/W = 10^{-4}, for the maneuver to be considered impulsive. Note that there is essentially no difference in transfer times among the various startup altitudes.

Figure 3.6 illustrates the variation in transfer time with vehicle thrust-to-weight. Note that LEO-GEO transfer time increases logarithmically as thrust-to-weight decreases below approximately 0.01. Also, transfer times are quite insensitive to start altitude variations between 185 and 1000 km.

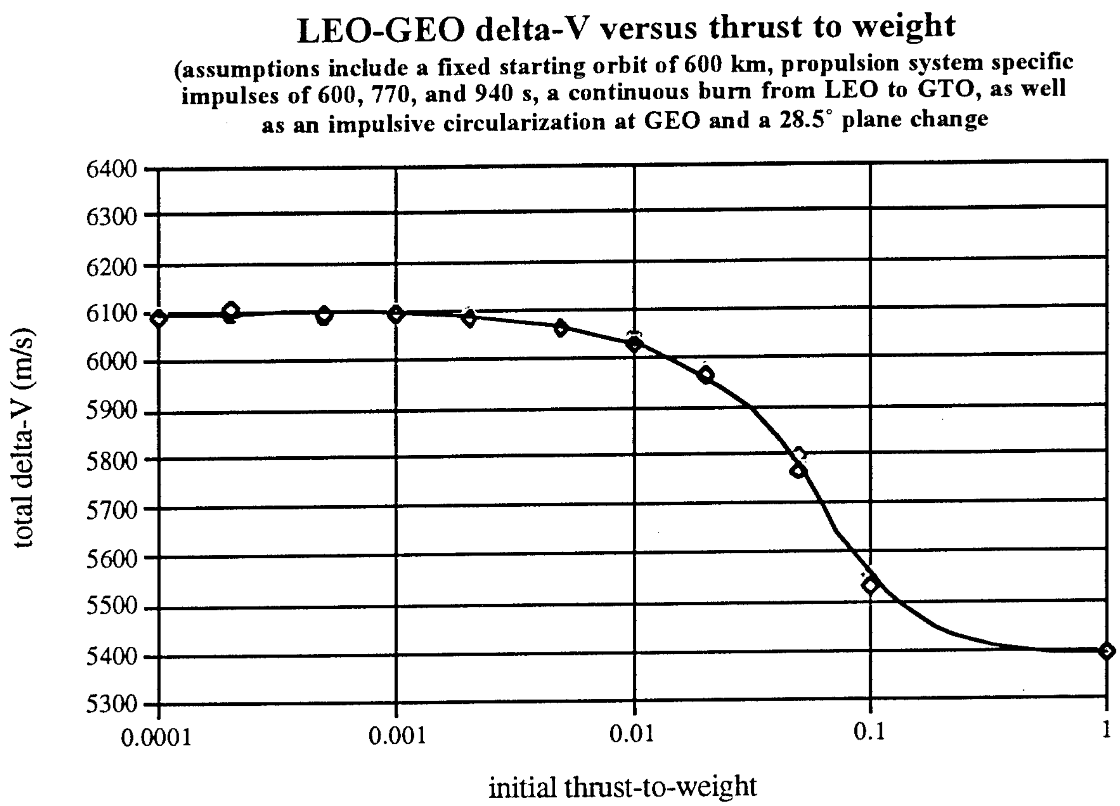


Figure 3.7 Assumptions are the same as in the previous figure (28.5° inclination, circular orbit).

Three specific impulse levels were examined: (1) *600 s*, requiring an H_2 exhaust temperature of 1215 K and an ideal expansion to vacuum; (2) *770 s*, approximating a 2000 K exhaust, and (3) *940 s* (2985 K). These correspond roughly to (1) the performance of a thermoelectric reactor system (e.g. SP-100) modified to perform propulsive maneuvers, (2) a thermionic bimodal system, and (3) a high performance nuclear thermal propulsion device based on a fuel form such as that used by the Particle Bed Reactor. The choice of hydrogen as a propellant is not arbitrary; see Appendix A for further details. Figure 3.7 shows the insensitivity of the LEO-GEO transfer delta-V to propulsion system specific impulse over the range of thrust-to-weight studied. Since delta-V is only indirectly affected by the system's I_{sp}, this is not a surprising finding. However, it does simplify the amount of work necessary to determine the optimal system.

Note the delta-V plateau between thrust-to-weights of 10^{-4} and 10^{-2}. A MILSTAR-class payload can be delivered to GEO by a 5 N thruster with the same propellant mass as a 500 N system. Of course, there is a significant increase in transfer time at the lower thrust-to-weight (in this case, 39 days at 5 N as opposed to 13 hours at 500 N).

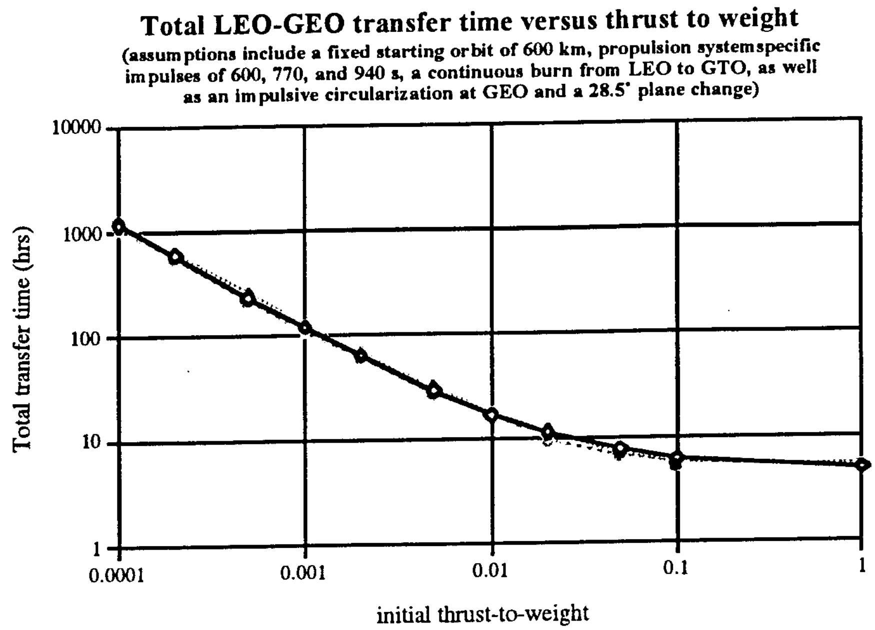


Figure 3.8 Assumptions are the same as in the previous figure.

In Figure 3.8, it is clear that specific impulse does not have a large effect on LEO-GEO transfer time. At higher I_{sp}, the system 'burns' less propellant for a given initial mass in low earth orbit; thus, the propulsion system is always propelling a larger weight at higher I_{sp}. This decreases average thrust-to-weight and increases transfer time very slightly.

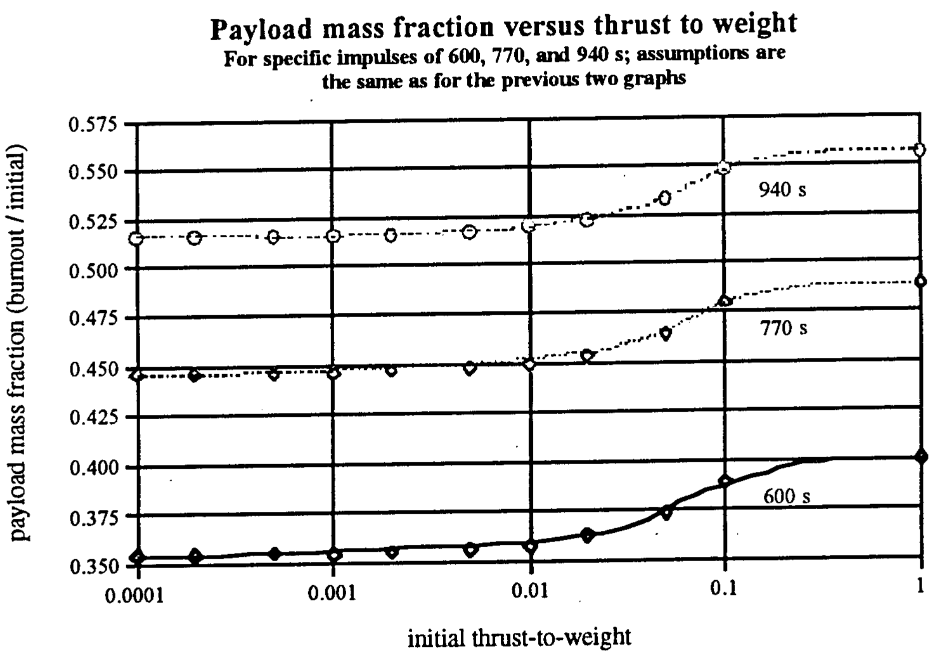


Figure 3.9 At 600 km, the LEO-GEO transfer delta-V is between 5400 and 6100 m/s, depending on thrust-to-weight. A 600 s propulsion system is capable of delivering payload mass fractions to GEO no greater than .40, while a high-performance system might achieve as much as .55.

A bimodal system would be capable of delivering approximately 37% of its initial mass in low earth orbit to GEO for the lowest level of specific impulse considered (Fig. 3.9). More than half of the initial mass can be delivered by the 940 s system. This directly affects the launch vehicle trades exhibited in the next set of figures.

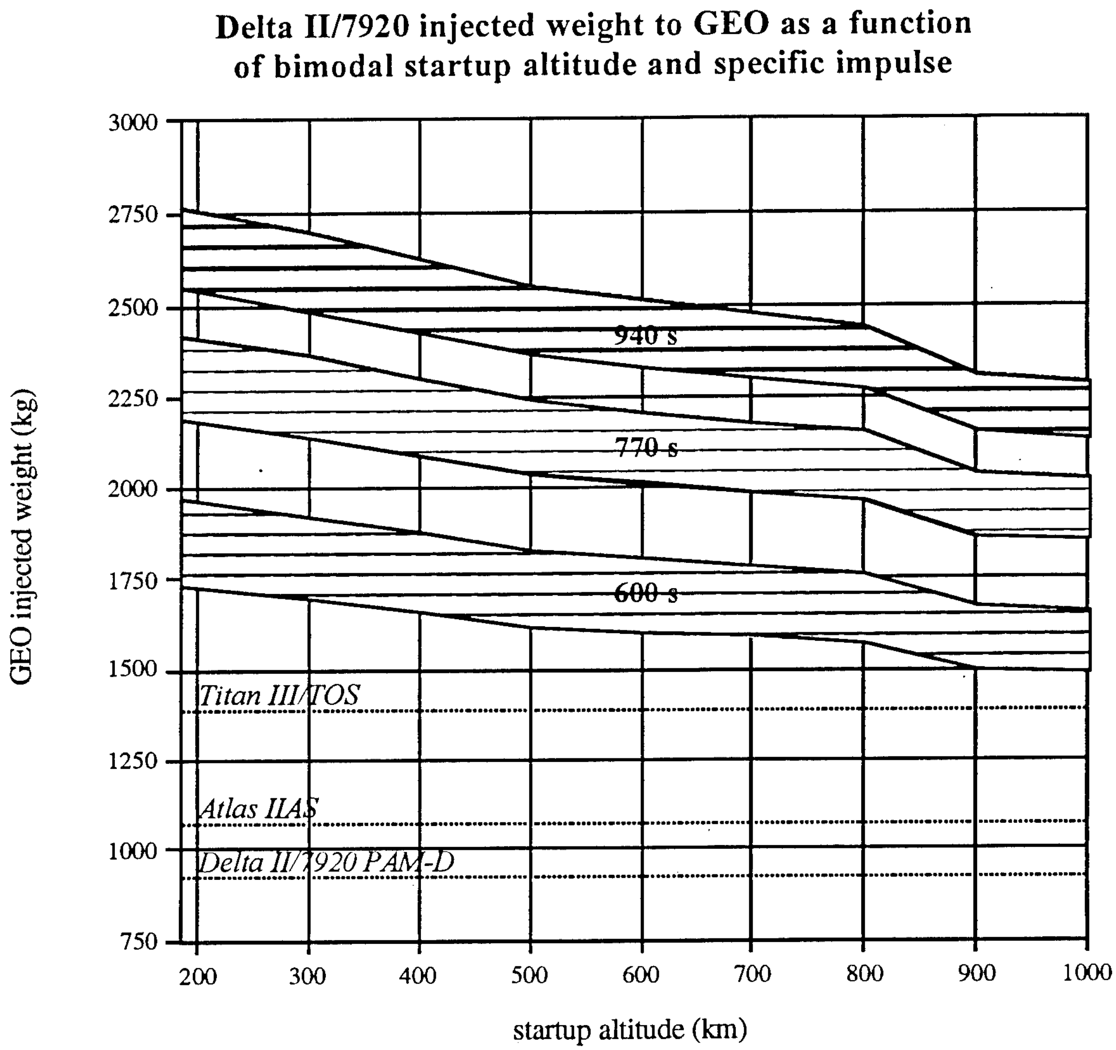


Figure 3.10 The Delta II/7920 payload capability to low altitudes is fairly flat. The broken lines for Delta II/7920 PAM-D, Atlas IIAS, and Titan III/TOS represent current maximum payload deliveries to GEO. The range of injected weights for a given altitude and Isp correspond to the performance of low thrust-to-weight (~ 10^{-4}) and high thrust-to-weight (~ 1).

In Figure 3.10 above, *injected weight* refers to the total payload mass placed on-orbit by the booster and bimodal system. The bimodal system is part of the payload, since it will continue to function as a power source and propulsion system once the operational orbit is reached.

Delta II 7920 has a very flat payload capability to low earth orbit; it retains 80% of its maximum capability at 1000 km. Even at the lowest I_{sp} studied (600 s) and highest altitude, the Delta/Bimodal combination places at least 1500 kg in GEO. This surpasses both Atlas IIAS and Titan III/TOS, the two next more capable systems. A high-I_{sp}, high-thrust system is capable of delivering 2750 kg to GEO.

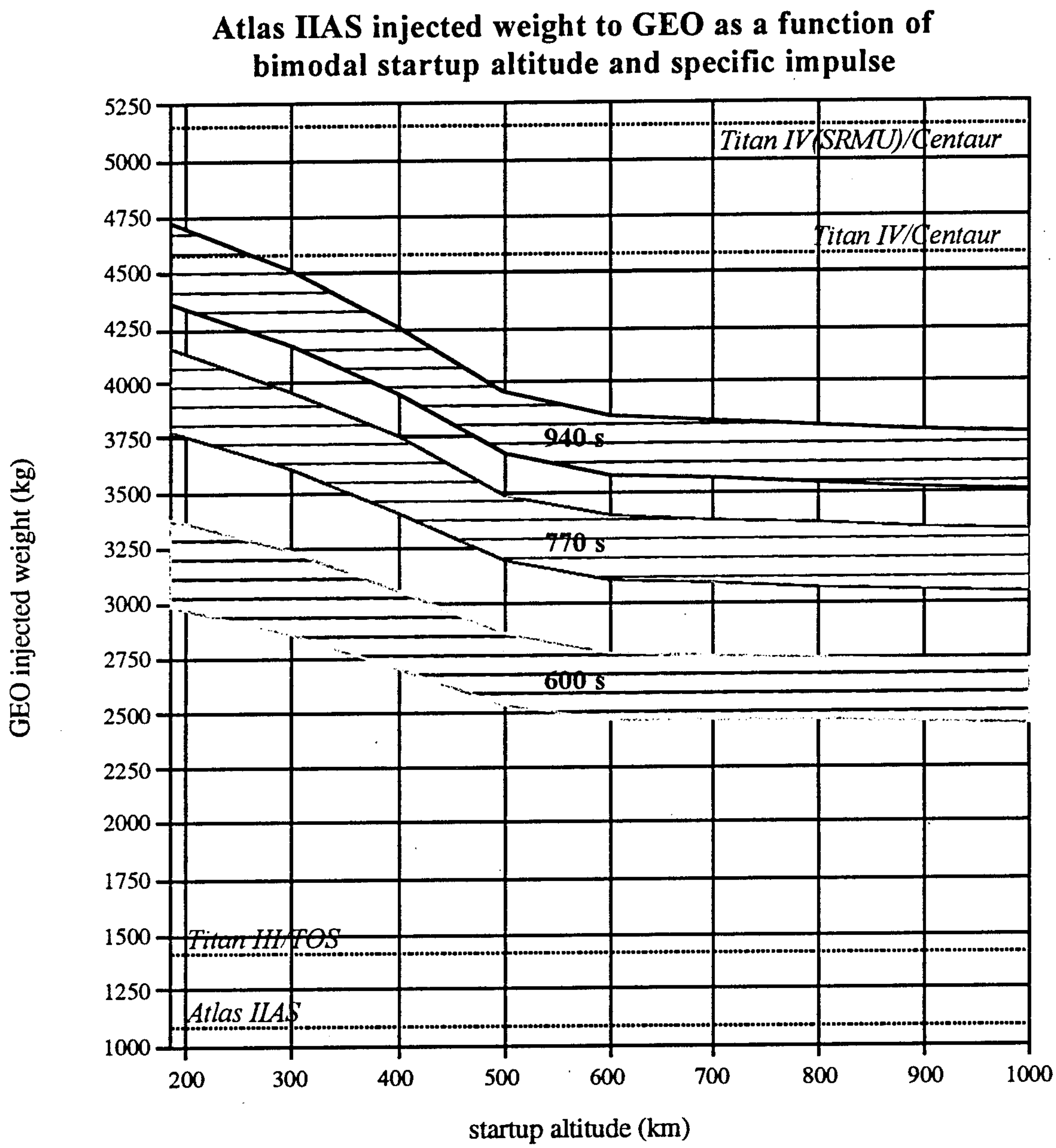


Figure 3.11 The range of injected weights for a given altitude and Isp correspond to the performance of low thrust-to-weight (~ 10^{-4}) and high thrust-to-weight (~ 1).

An Atlas IIAS/Bimodal combination is only capable of placing Titan IV/Centaur (4540 kg) payloads in GEO if the bimodal system is designed to operate at high specific impulse

and can fire at low altitudes--under 250 km (Fig. 3.11). The Atlas IIAS is not capable of stepping down a Titan IV(SRMU)/Centaur payload. However, a low-I_{sp} system can

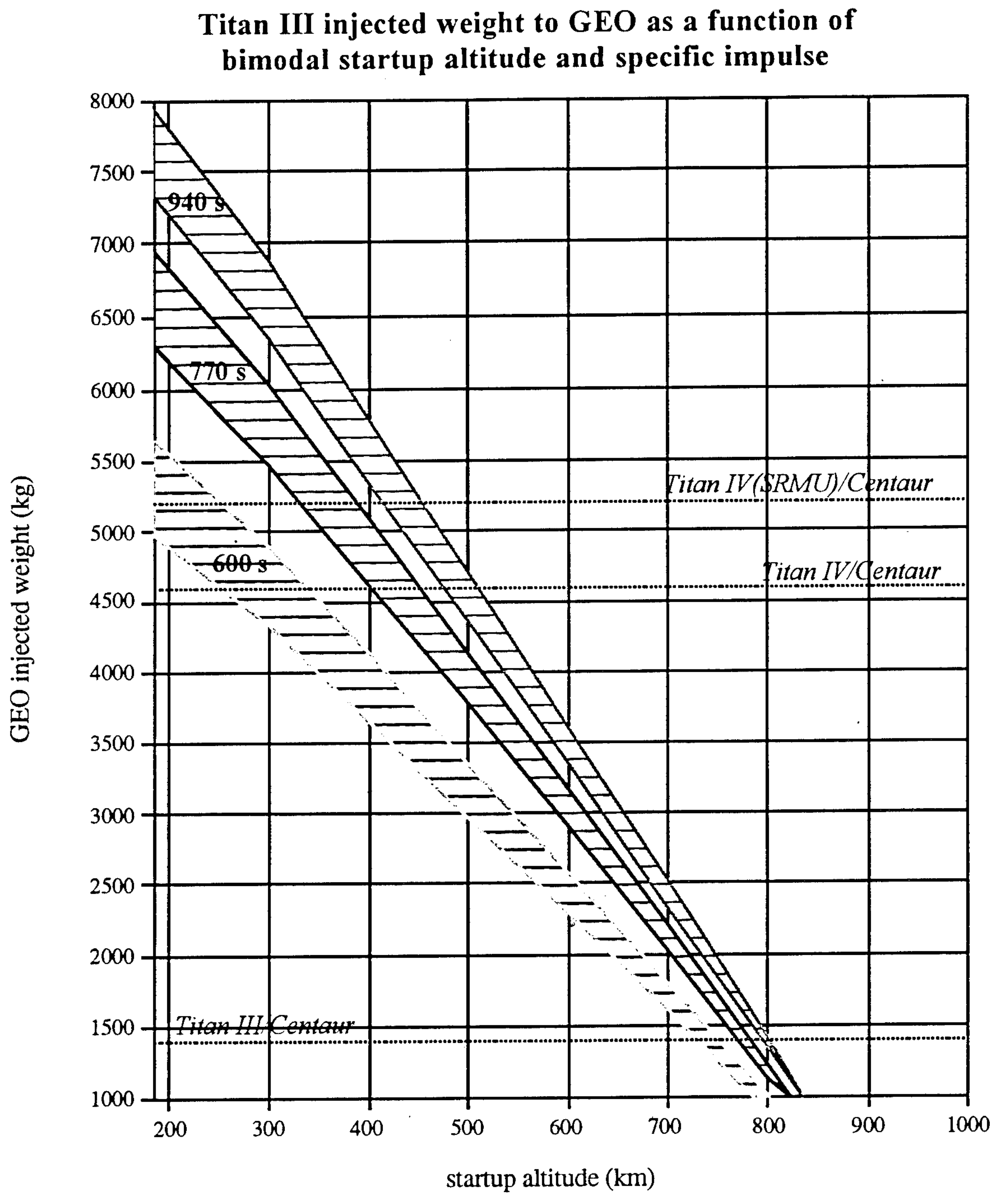


Figure 3.12 The range of injected weights for a given altitude and Isp correspond to the performance of low thrust-to-weight (~ 10^{-4}) and high thrust-to-weight (~ 1).

achieve a minimum of 2500 kg to GEO; a system operating at 770 s can place payloads that are 93% as massive as that achieved by Titan IV/Centaur. This stepdown is

achievable by using an electric propulsion system with a specific impulse of 1000 s or greater; however, trip times can be two years or longer, depending on system power and specific impulse [Ref. 6].

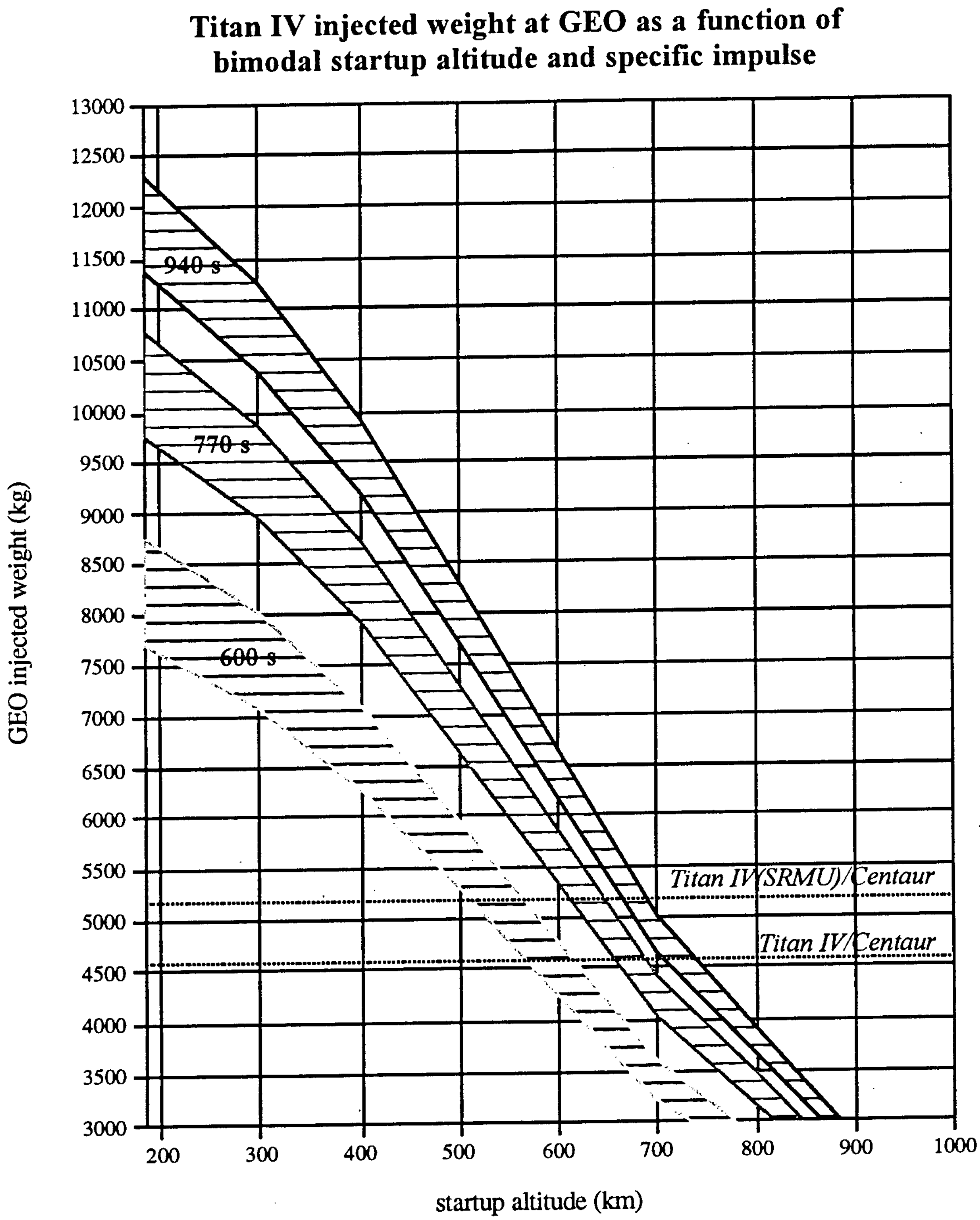


Figure 3.13 The range of injected weights for a given altitude and Isp correspond to the performance of low thrust-to-weight (~ 10^{-4}) and high thrust-to-weight (~ 1).

A Titan III/Bimodal combination breaks even with its conventional counterpart at initial bimodal firing altitudes of between 750 and 800 km (Fig. 3.12). In order to achieve Titan IV-class payloads, the medium-Isp (770 s) system must fire at altitudes under 450 km; a Titan IV(SRMU) payload can be delivered if the firing altitude is lowered to ~ 350 km. At minimum altitude, even the 600 s system surpasses the Titan IV(SRMU)/Centaur performance.

Thrust level requirement for various LEO-GEO transfer times from 1 day to 4 weeks (GEO payload masses of 2300, 3900, 6500, and 10250 kg)

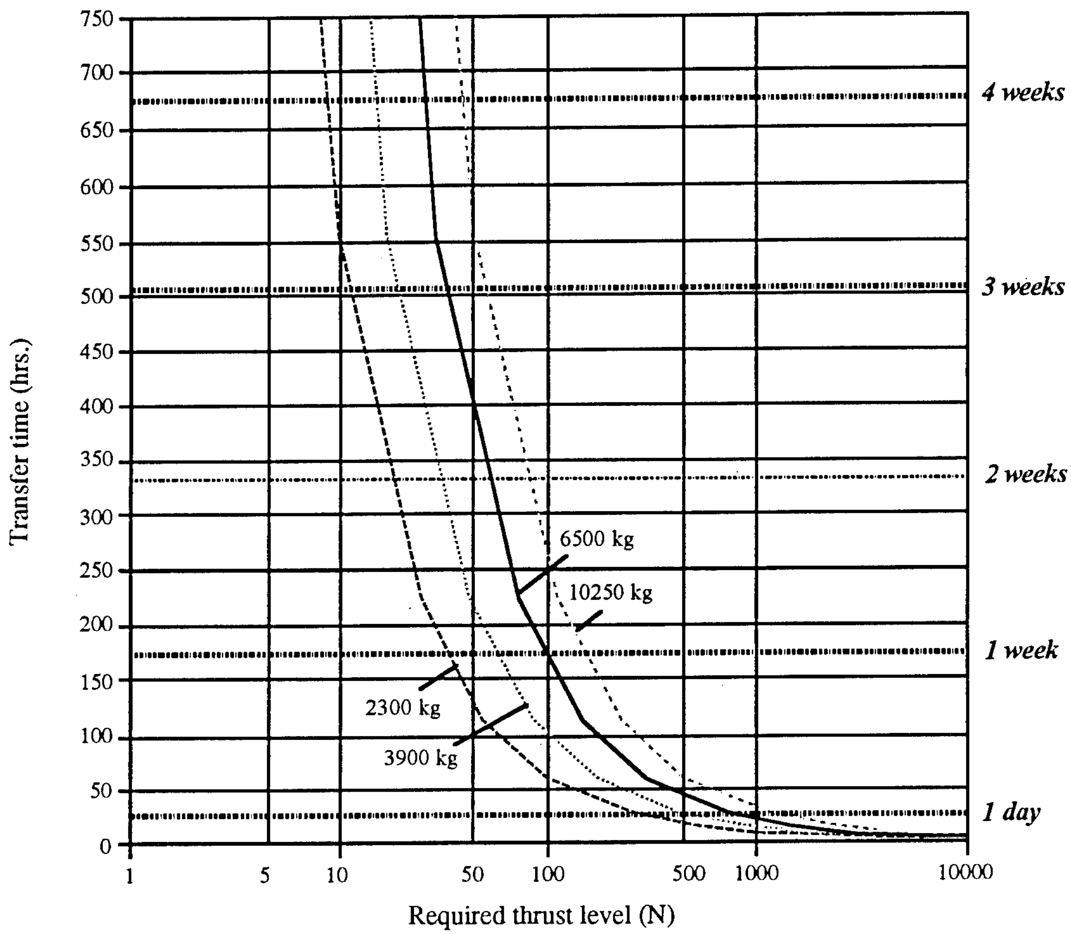


Figure 3.14 Transfer times of between one day and four weeks are examined. Note that a 770 s bimodal system coupled to a Titan IV(SRMU)/Centaur can place approximately 10250 kg in GEO; if this payload must be operating on-orbit within one week, the minimum bimodal system thrust must be 160 N. To achieve a transfer within a day, this requirement jumps to over 1000 N.

Breakeven for a Titan IV/Bimodal system occurs at altitudes of between 600 and 750 km, depending on specific impulse and vehicle thrust-to-weight (Fig. 3.13). Low-altitude

firings allow payload deliveries of 7500 to 12500 kg; the Russian Energia is capable of 18000 kg [Ref. 7]. At 770 s and an initial altitude of 185 km, a bimodal system enables Titan IV to place a Direct Broadcast HDTV satellite in GEO.

The bimodal system thrust level is determined by the maximum GEO payload and the maximum transfer time requirement for that payload. Currently, Titan IV(SRMU) /Centaur provides the greatest payload capability available (5200 kg); with a one-week transfer time requirement, the bimodal device must deliver ~ 80 N of thrust. This declines to 20 N if the transfer time requirement is relaxed to four weeks. Figure 3.14 illustrates the needed thrust given a range of transfer times from 1 to 28 days.

Table 3.3 shows the launch cost savings associated with the various possible stepdowns. A complete cost analysis is beyond the scope of this report. In order to detail the actual cost savings associated with stepdown, it would be necessary to quantify the costs of (1) individual satellite power and propulsion systems, and remove them; (2) bimodal system development; (3) any additional launch and operations infrastructure needed for a bimodal system; and (4) the bimodal power and propulsion by unit.

Stepdown Type	*Launch Cost Savings (FY92$)*	*Bimodal Requirements*
Titan IV/Centaur to Titan III/Bimodal	**$255M**	<475 km at 940 s <400 km at 770 s <250 km at 600 s
Titan IV/Centaur to Atlas IIAS/Bimodal	**$280M**	Achievable only at 940 s, <250 km
Atlas IIAS to Delta II 7920/Bimodal	**$70M**	Achievable with all systems examined at all altitudes up to 1000 km

Table 3.3 Bimodal System Requirements to Achieve Launch Vehicle Stepdown
(assumptions for launch costs do not include substitution of the bimodal system for the satellite's integral power and propulsion systems, development costs, or costs associated with launch infrastructure at either of the two major US facilities--ETR and WTR. This is a strict booster cost trade, with cost figures coming from the JPL Launch Vehicles Summary for Mission Planning, JPL D-6936 Rev. C.)

It is possible to achieve three stepdowns: Titan IV/Centaur (*T4/C*) to Titan III, Titan IV/Centaur to Atlas IIAS, and Atlas IIAS to Delta II 7920. It is possible to move *T4/C* payloads (4560 kg) to the Atlas booster only if a high thrust-to-weight, high-I_{sp} bimodal system is used at low start altitudes. However, there appears to be only marginal utility to this particular stepdown, as the move from *T4/C* to Titan III provides 90% of the launch cost savings. A low-performance bimodal system (600 s I_{sp}) can perform this

stepdown but only at very low start altitudes. The Atlas-Delta move is possible with any of the systems examined.

Satellite	***Remarks***
MILSTAR	*Stepdown* from *T4/C* to Titan III or Atlas IIAS
FEWS	*Stepdown* from Atlas IIAS to Delta II 7920 and a *770 s* bimodal system
Direct Broadcast HDTV	*Enabled* by *T4/C* and a *770 s* bimodal system

Table 3.4 Performance Gains for GEO Satellite Systems

MILSTAR, which currently is baselined for launch aboard a Titan IV/Centaur, could be moved down to Titan III or Atlas IIAS when these smaller launchers are used in conjunction with a bimodal power and propulsion system. This potentially saves $255-280M in launch costs (Table 3.4). Likewise, the FEWS satellites can be moved from Atlas to the smaller Delta II/Bimodal combination, for a savings of $70M. To achieve this stepdown requires an I_{sp} of 770 s and a maximum startup altitude of 700 km. Finally, the Direct Broadcast HDTV satellite is enabled by a Titan IV with a 770 s bimodal system firing at altitudes of between 185 and 250 km. At 10 metric tons, the specific cost of delivering the DB HDTV satellite is only $27,500/kg. This assumes a one-for-one substitution of the bimodal system for both a solar photovoltaic/battery power system and onboard hydrazine thrusters; it cuts Titan IV's cost/kg by 65%.

While it is clear than stepdowns are enabled by the systems examined, it is important to note that extra payload margin also provides a dual-satellite launch capability. This is seen to be possible for all boosters analyzed, although the requirements that must be placed on the bimodal system vary. One difficulty that arises when multiple payloads are launched from the same booster is the payload shroud volumetric constraint. Liquid hydrogen--the bimodal system's propellant--has a density of only 70 kg/m^3, fourteen times lighter than water. Appendix B details these problems.

These facts dictate the following bimodal system requirements:

(1) A propulsion system specific impulse of *770 s*.

(2) A thrust level of *80 N* in order to achieve 7-day transfer times for MILSTAR-class payloads; these thrust levels allow larger systems such as the DB HDTV satellite (10000 kg) to be moved to GEO in ~ 2 weeks.

(3) A 28-day transfer time requirement would reduce this thrust level specification to *20 N*.

3.2 Sun-Synchronous Orbit Repositioning

Satellites in sun-synchronous orbit (SSO) pass over the equator at the same local time on every orbit. This enables a satellite to have a constant viewing angle over the target area. which is extremely useful is for weather and other observation platforms. For instance, a system in SSO could reside at an altitude of 882 km and have an equatorial overflight time of 4:15 PM. The satellite would be said to be in a *4:15 orbit*; on each of its 14 passes per day, the local (ground) time would be approximately 4:15 PM. Note that the satellite will also pass over the night side of the planet's equator at a local time of 4:15 AM. Sun-synchronous orbits can be circular or elliptical; to gain higher resolution, some systems might drop down to fairly low perigee altitudes.

Sun-Synchronous Orbit

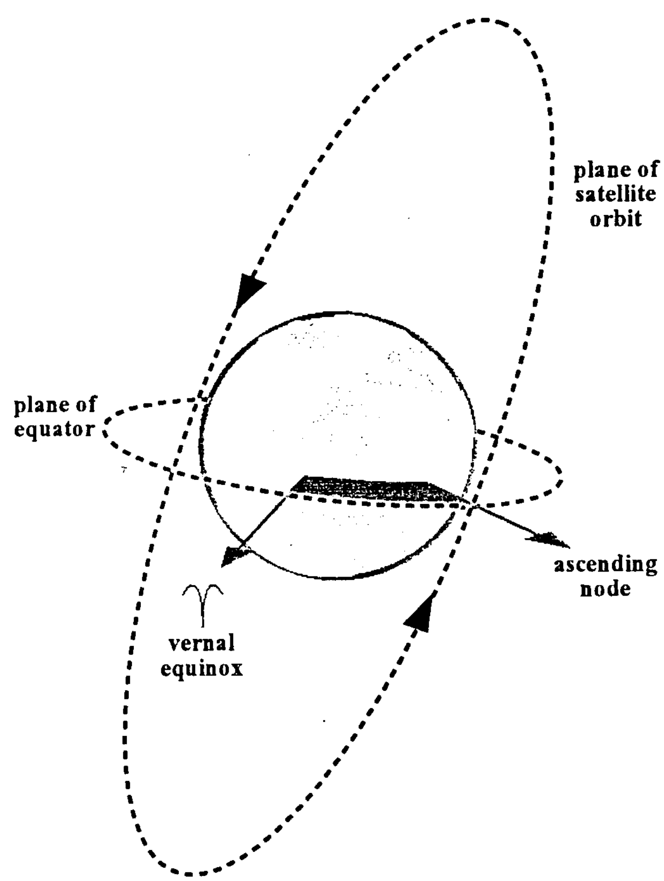


Figure 3.15 The geocentric angle between an arbitrary fixed reference (in this case, the vernal equinox) and the ascending node of the orbit is the right ascension of the ascending node (RAAN). Allowing the rate of change of the RAAN to equal the shift in the sun's position relative to the vernal equinox maintains a constant viewing angle and a constant equatorial overflight time.

To achieve these constant viewing angles requires control over the right ascension of the ascending node (RAAN); essentially, RAAN is the geocentric angle measured from an inertial reference (in this case, the vernal equinox or *first point of Aries*) to the point where the satellite passes through the plane of the ecliptic, from the southern to northern hemispheres [Ref. 8, pp. 123-5]. In order for a satellite to maintain a constant overflight time, it must match the earth's revolution about the sun (.9856°/day). By varying the inclination of the satellite's orbit, it is possible to do this; there are a number of perturbations (e.g. the sun, moon, and shape of the earth) that effect the rate of change of a particular platform's RAAN. At 555 km, for instance, an inclination of 97.6° is required to achieve the .9856°/day RAAN shift. Figure 3.15 is a pictorial representation of RAAN.

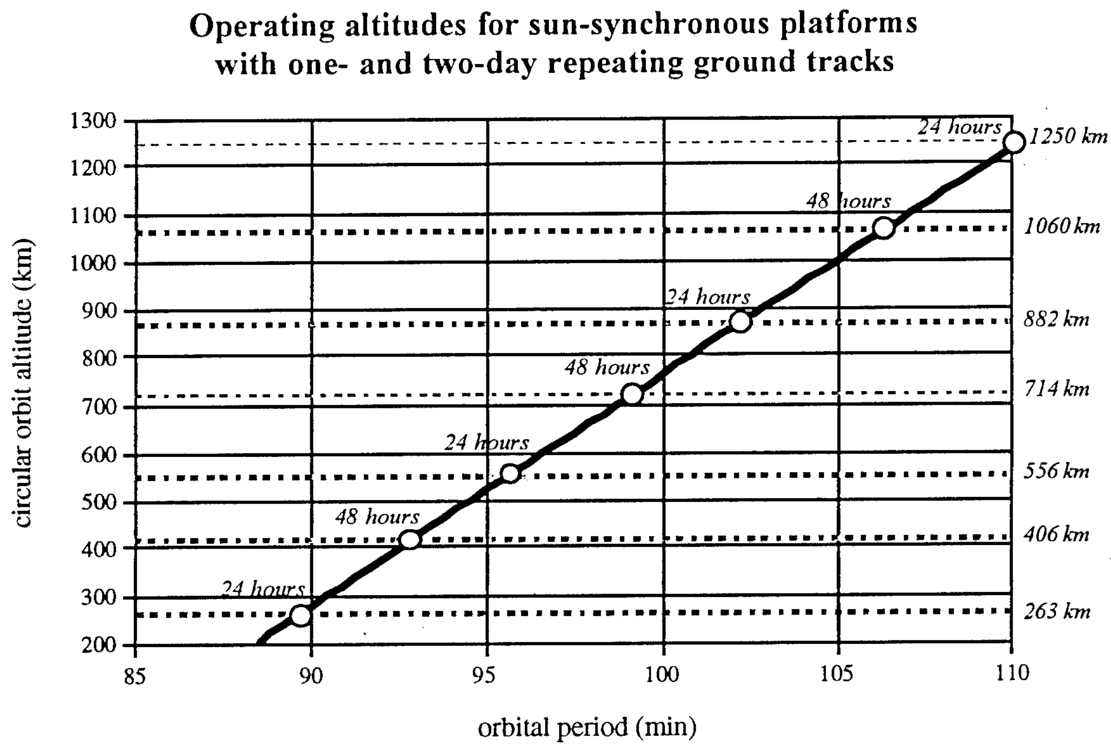


Figure 3.16 In order for the ground track to repeat every day, the length of the day must be a precise multiple of the orbital period; the satellite makes an integer number of orbits in 24 hours. This is shown for 16 orbits/day (89.75 min. period), 15 orbits/day (95.73 min.), 14 orbits/day (102.57 min.), and 13 orbits/day (110.46 min.). Achieving these orbital periods requires specific circular orbit altitudes, as shown. For two-day repeats, the satellite must make an integer number of orbits in precisely 48 hours.

Achieving a repeating ground track (such that the trace of the satellite falls over the same path after a specific number of orbits) necessitates the choice of specific altitudes. Figure 3.16 shows a series of operating altitudes that have one- and two-day repeating ground tracks. Ideally, a ground track should repeat quickly for systems that require updated information about specific regions of the globe. Depending on the capabilities of the sensing instruments, most notably the swath coverage of the instrument, it might be desirable to go to longer cycles in order to achieve higher-resolution data (at the cost of longer wait times until a particular target is in view).

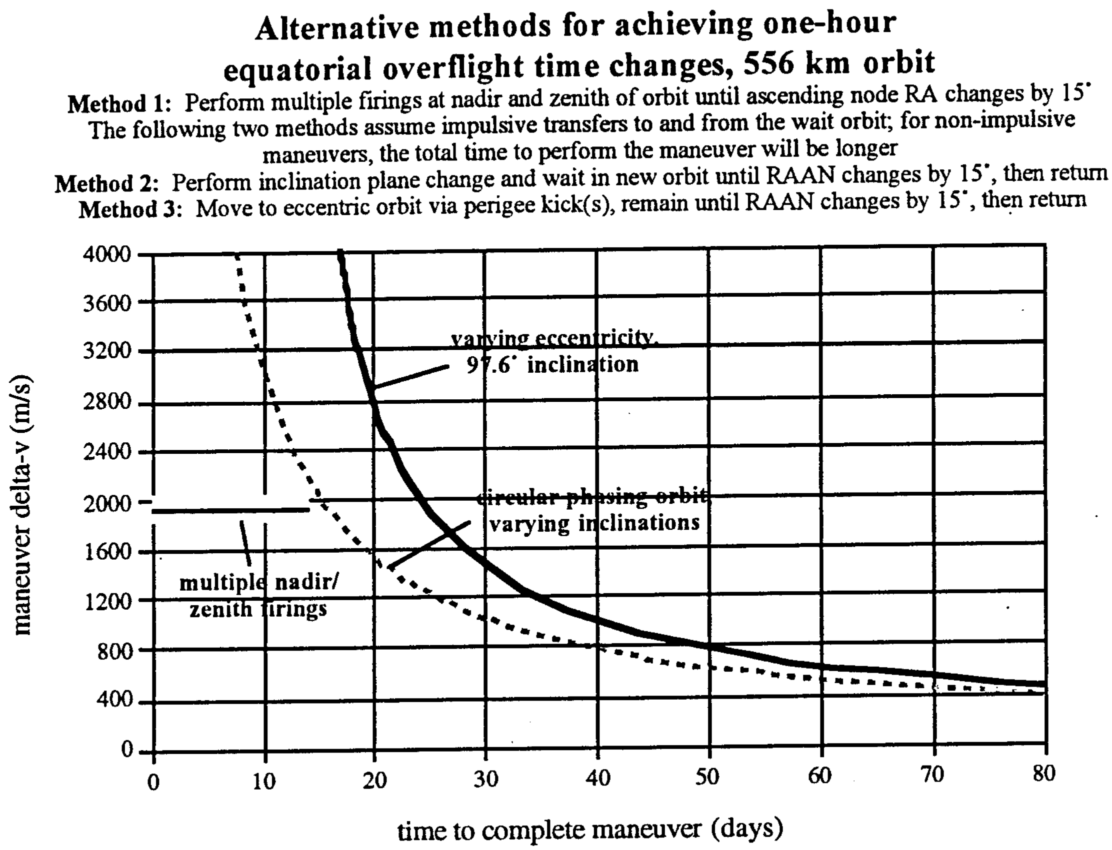


Figure 3.17 Three methods of achieving overflight time changes.

Once a system is locked into a specific overflight time, it is difficult to modify it. It is conceivable that a satellite platform with a *4:15 orbit* is not collecting the type of data its users had originally intended; or perhaps, in the case of reconnaissance platforms, the satellite's orbit might be well-known to potential adversaries who can make preparations to hide their intentions and capabilities during the satellite's pass. By changing the overflight time to, say, *5:15*, the system could now collect information that it was previously incapable of acquiring. The only alternative would be to fly a second satellite

in a *5:15 orbit*. Figure 3.17 illustrates the total transfer time required for three types of maneuvers that would permit a one-hour change in overflight time.

Note that none of these are inexpensive in terms of propellant. The quickest method, which involves firing the thrusters successively at the nadir and zenith of the orbit, requires between 1800 and 2100 m/s (this is essentially a plane change) for altitudes of up to 1250 km. At a thrust level of 50 N, a 5000 kg-class satellite would need approximately two weeks to perform the needed burns to achieve a one-hour change in overflight time (Fig. 3.18). At 500 N, this maneuver requires only 35 hours.

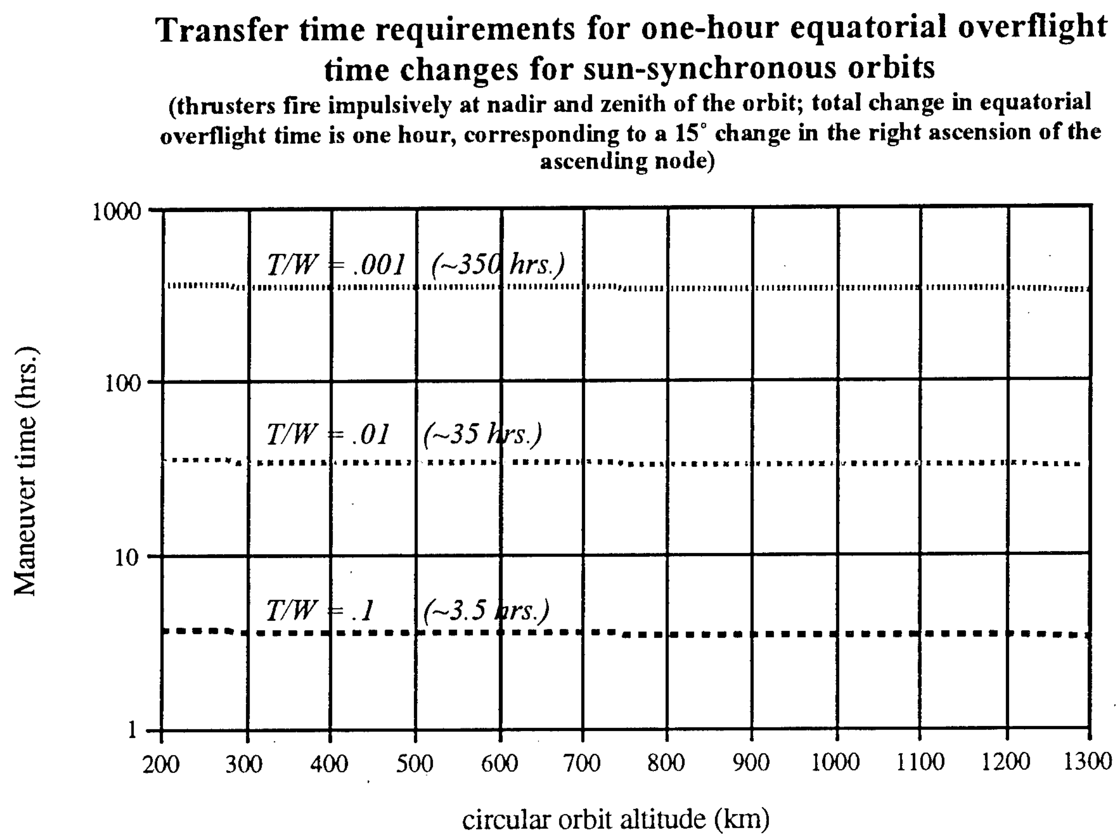


Figure 3.18 The time required for nadir/zenith firings is heavily dependent on thrust-to-weight.

Two other methods provide a means to perform these maneuvers at decreased propellant cost. By varying either the inclination or eccentricity of the satellite orbit through thruster firings, the right ascension of the ascending node can be made to shift at a rate other than the required .9856°/day needed to hold position. When the desired amount of RAAN change is achieved, the inclination/eccentricity is restored by a second series of burns. Increasing the eccentricity was shown to be a less efficient maneuver than modifying the inclination of the orbit. It is also possible to perform a continuous burn that would combine these two effects: thrusting near the nodes of the orbit changes the

inclination plane, while thrusting near the nadir and zenith of the orbit modifies the RAAN. While this is not examined in detail, it might provide a compromise between transfer time and propellant usage.

Propulsion System	Fast Transfer (2000 m/s, 1-15 days)	Slow Transfer (400 m/s, 80 d)
Bipropellant hydrazine	2587 kg	678 kg
LOX/LH_2	1807 kg	429 kg
Direct Nuclear Thermal	976 - 1442 kg	213 - 329 kg

Table 3.5 Propellant Mass Requirements for Fast and Slow SSO Maneuvering (assumes a 5000 kg-class satellite platform in a 556-km circular orbit, a 15° RAAN change (one hour overflight time change), a bipropellant hydrazine I_{sp} of 280 s, LOX/LH_2 I_{sp} of 455 s, and a Direct Thermal I_{sp} of 600-940 s)

There are three identifiable options for performing this type of maneuver: (1) a ***fast*** (hours to days) transfer consisting of multiple nadir/zenith firings that requires a total delta-V of approximately 2000 m/s, (2) a ***slow*** transfer consisting of an inclination plane change which could cost as little as 400 m/s if 80 days or more of wait time are available to the user, and (3) a compromise transfer that was not analyzed in this report, which uses continuous thrusting to perform both inclination and RAAN change (Options 1 and 2 provide performance boundary conditions for this type of maneuver). The second option allows virtually any change in overflight time, if the satellite remains in its phasing orbit; the first option allows only a one-hour change. To achieve larger RAAN shifts during fast transfers will require a larger delta-V--and the relationship scales linearly. Table 3.5 illustrates the propellant cost for these maneuvers aboard a MILSTAR-class platform.

Notice that over half of the initial mass of the satellite is necessary in order to perform the fast maneuver with conventional bipropellant hydrazine thrusters; for the slow maneuver, over 600 kg is needed. Thus, such missions could be performed with present technology if the user is willing to sacrifice a large fraction of dry payload on-orbit. Moving to a direct thermal system would lower this by approximately a factor of two in both cases. Electric propulsion systems cannot be used for this maneuver due to the high thrust-to-weight needed for impulsive maneuvers (i.e. short-duration firings at the nadir and zenith, for instance).

The thrust levels required to achieve various transfer times are shown in Table 3.6 for a propulsion system I_{sp} of 770 s. Note that, at the altitude studied, this allows a total of 228 minutes of impulsive thrusting per day. For the slow transfers, it is assumed that all

thrusting (both transfer to and return from the wait orbit) is performed in one day; therefore, 114 minutes are available at the beginning and end of the slow maneuver for thrusting in order to satisfy the impulsive approximation.

Maneuver	*2000 kg platform*	*5000 kg platform*	*10000 kg platform*
Fast 1-day	257 N	643 N	1285 N
Fast 7-day	37 N	93 N	185 N
Slow 28-day	155 N	387 N	774 N
Slow 80-day	58 N	146 N	292 N

Table 3.6 Thrust Level Requirements for SSO Maneuvering
(assumptions include a 556-km circular orbit, 97.6° inclination, a fast transfer delta-V of 2000 m/s, a slow (28-day) delta-V of 1100 m/s (550 m/s initial, 550 m/s final burn), an 80-day delta-V of 400 m/s (200 m/s initial, 200 m/s final burn), and a system specific impulse of 770 s; a system at this altitude completes 15 orbits per day)

Most of these thrust levels are higher than those required by the LEO-GEO transfers examined in the last section.

3.3 GEO Repositioning

Satellites in geosynchronous orbit are by definition restricted in their coverage to that portion of the earth's surface within their field of view. For instance, a system residing at 75° W longitude would view approximately one-third of the earth; this position provides good coverage of the eastern seaboard of the United States. This system is therefore capable of viewing the area between approximately 135° W and 15° W longitude, with poorer coverage at the poles and limbs.

Full earth coverage can theoretically be provided by three satellites with some overlap at the margins. In some cases, coverage by a single system does not satisfy user requirements and a second satellite must be launched or repositioned on-orbit to augment capabilities. Gen Horner, CINCSPACE, noted that he would require stereo coverage of Bosnia-Hercegovina in order to conduct an air war in that region. "We have extra DSP satellites and we have extra launchers," Horner stated, "but when my people tell me it takes a year to put one on station, this fighter pilot has trouble with that." [Ref. 9]. While Gen Horner's proposed solution to this was to advocate quick-launch capability, repositioning satellites already on-orbit might provide an alternative to acquiring a new launch system. These satellites could then return to their normal duties following a critical operation.

Repositioning of satellites is currently done in response to a need during a crisis (e.g. the Gulf War, a possible US-led response in Bosnia). The satellite fires its thrusters and

inserts itself into an elliptical transfer orbit (ETO) having a period somewhat longer or shorter than the standard 1436-minute (one-day) circular orbit and then circularizing the orbit at GEO after the required longitudinal shift was effected. For example, moving to a 1324-minute ellipse would require a perigee-lowering burn that would place the satellite at its original position 12 minutes sooner than it would have if it had remained in GEO. Circularizing upon return to apogee--that is, geostationary altitude--would effectively place the satellite 3° east of the original position. Clearly, any longitudinal shift can therefore be accomplished if the satellite user is not operating under a time constraint; at a 3°/day drift rate, a satellite stationed above the US East Coast could be moved directly over the Persian Gulf region in 43 days. 3°/day is representative of current operational capabilities.

GEO repositioning delta-V requirements versus longitudinal rate of change (degrees/day)
(assumptions include the use of an impulsive transfer to an elliptic orbit of longer or shorter period and an impulsive return to GEO)

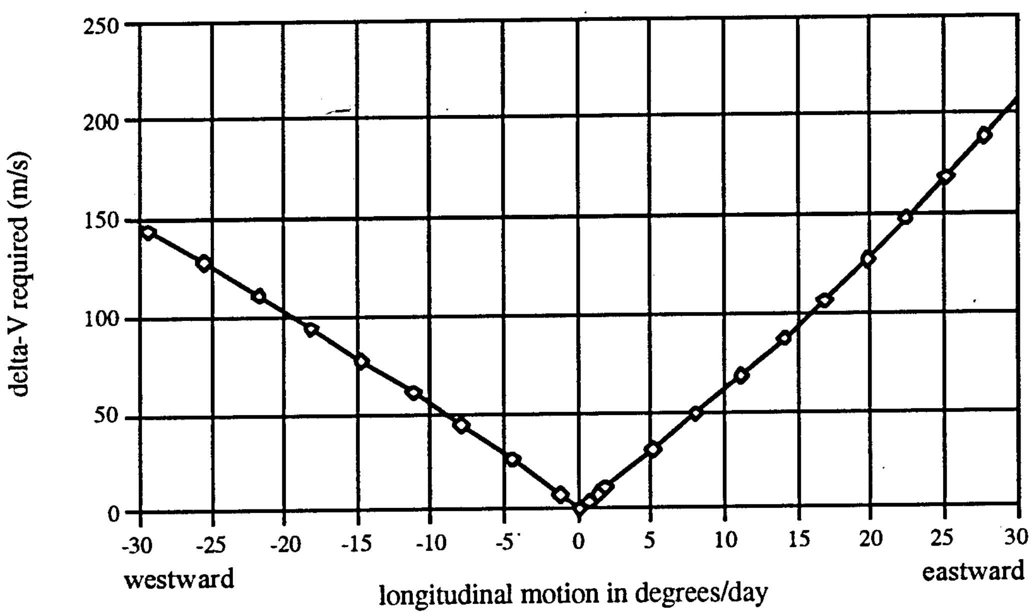


Figure 3.19 Moving westward requires an insertion into an orbit with an apogee higher than GEO; moving east is performed by lowering the perigee of the orbit below GEO, which is less expensive.

Higher longitudinal drift rates increase system responsiveness by allowing greater satellite mobility; however, responsiveness is improved only at the expense of fuel. Figure 3.19 illustrates the cost of these maneuvers in required velocity change. Increasing the drift rate from 3° to 30°/day would lower the transfer time between the East Coast and the Gulf from 43 to under 5 days, but would require between 7.5 and 10 times the propellant

expenditure (depending on the direction of travel). Note that a westward drift is always easier to accomplish than the corresponding eastward motion. This is because moving westward requires an apogee-raising burn--and the transfer to and from these orbits costs less in terms of delta-V than the comparable ETO needed to move east.

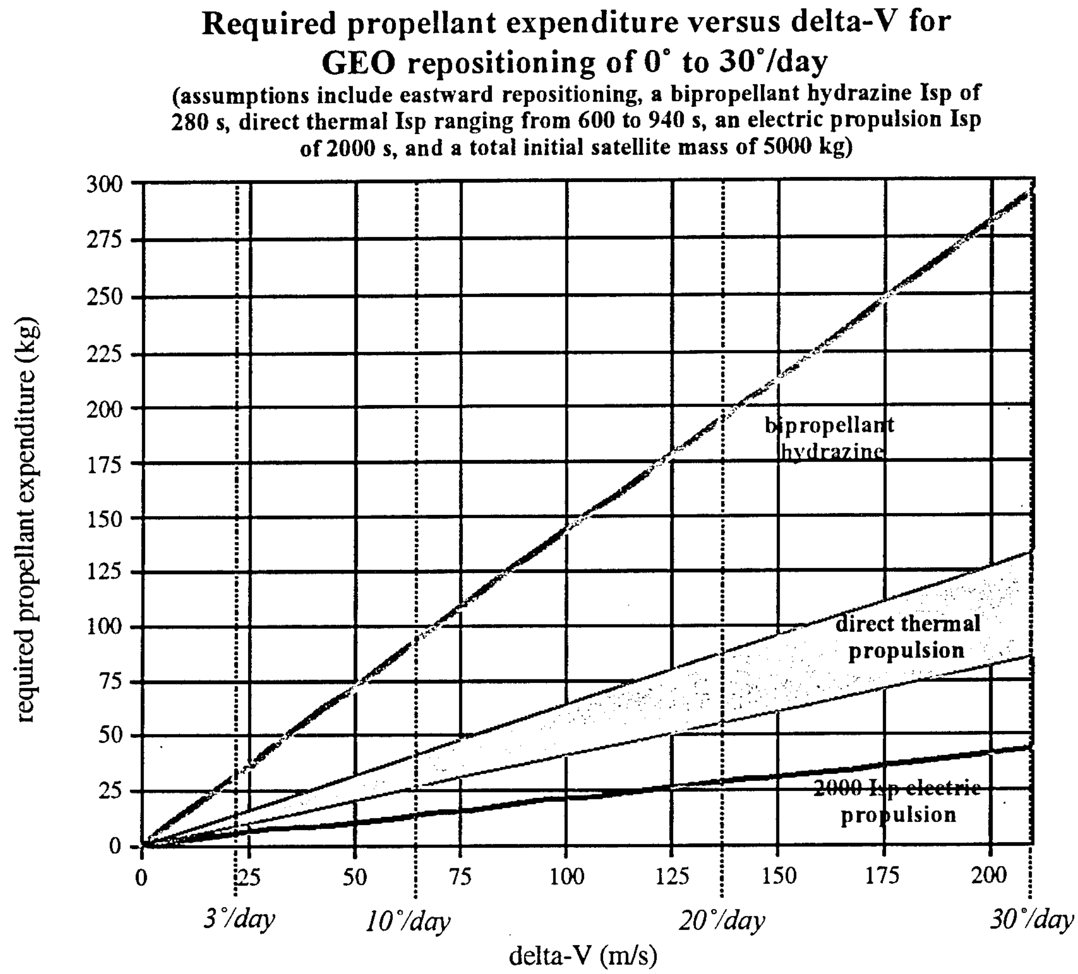


Figure 3.20 Direct nuclear thermal propulsion cuts propellant consumption by approximately a factor of three. Using high-performance electric propulsion (2000 s) would reduce overall consumption by a factor of six or more.

Figure 3.20 shows the propellant expenditure in kilograms for various drift rates. The satellite mass is 5000 kg, approximately that of a MILSTAR platform. Notice that, at 3°/day, the baseline system requires approximately 36 kg of propellant (a combustible hydrazine/nitrogen tetroxide mixture). At the same drift rate, the 770 s direct thermal system uses only 13 kg; a set of 2000 s electric thrusters would need only 5 kg.

At a 30°/day drift rate, the bipropellant scheme would need nearly 300 kg of fuel to perform the maneuver. This is greater than half of the total assumed propellant tankage

onboard MILSTAR (~500 kg). The direct thermal system requires 110 kg, while the electric system would need only 50 kg.

The electric system provides very low thrust and would not be capable of performing the impulsive (dual-burn) maneuver described above; however, by firing continuously throughout the transfer, electric thrusters acquire only a small delta-V penalty and essentially no transfer time penalty [Ref. 6]. Since thrust-to-weight is not a figure of merit for the drift rates studied, high-I_{sp} electric propulsion systems appear to be the optimal system choice for GEO repositioning.

3.4 High-Earth Orbit (HEO) Transfer

While geosynchronous orbit is arguably the most demanding of common orbital transfer missions, there are a number of other destination orbits between low earth orbit and GEO that provide useful locations for satellite platforms. The 24 NAVSTAR Global Positioning System (GPS) satellites are located in a circular 12-hour orbit, over 20000 km above the earth. This is somewhat more than half the distance to geosynchronous orbit, but will be referred to as *half-GEO*. They provide position, velocity, and time information on a worldwide basis to any user with GPS receiver equipment. The current GPS mission requires a Delta II-class launcher in addition to a 1000-kg apogee kick motor (AKM), needed to circularize the satellite's orbit. The total injected weight at half-GEO is ~2000 kg (AKM and satellite). Additional information is provided in Table 3.7.

A Global Air Traffic Control (GATC) system would solve the current problem of limited aircraft detection capabilities over ocean areas. The requirements for such a system are listed in Table 3.7, and are taken from a 1989 Lockheed report on space-based radar (SBR), the *SBR Nuclear Enhancement Study* [Ref. 10]. It is assumed that SBR requirements should be fairly similar to those for GATC. Lockheed examined several constellations, including an 12-satellite architecture at 1100 km to one residing at 2555 km and having only eight satellites. The latter constellation is not within present US capabilities; it would be necessary to use a Russian booster to achieve this orbit.

Satellite	Mass (kg)	Power (kWe)	Lifetime (yrs)	Launcher	Launch date	Constellation
GATC	*16000*	*45*	*10*	*NA*	*Post-2000*	*8 w/spare*
GPS	1000	~1	7	Delta II	Ongoing	24 w/3 spares

Table 3.7 Two HEO Satellite Platforms and their Characteristics
(NA: None Available in US)

Achievable injected weight for various delta-V requirements, Delta II 7920 booster

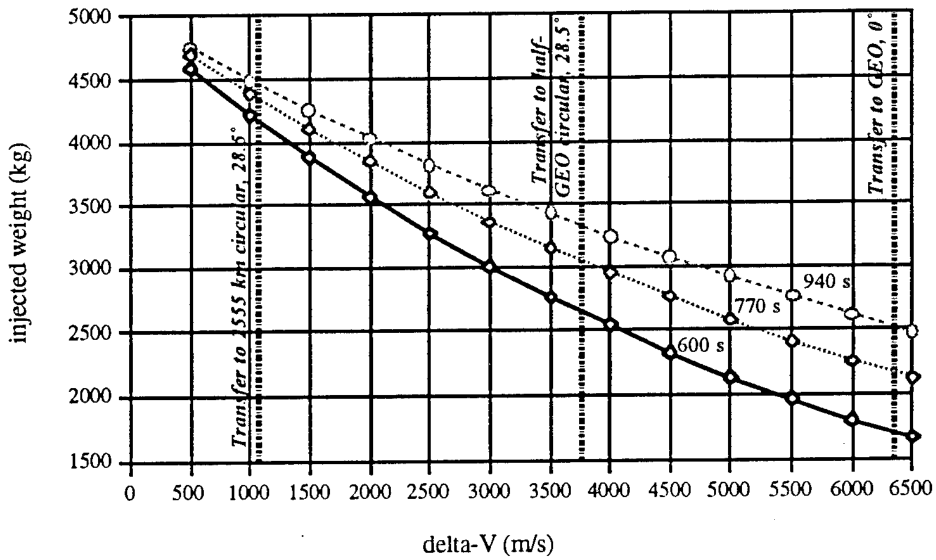


Figure 3.21 Assumptions for this chart include a launch from ETR into a 185-km circular parking orbit, a bimodal reactor system operating at a thrust-to-weight of 10^{-4}, and a variable I_{sp} (600, 770, and 940 s).

Achievable injected weight for various delta-V requirements, Atlas IIAS booster

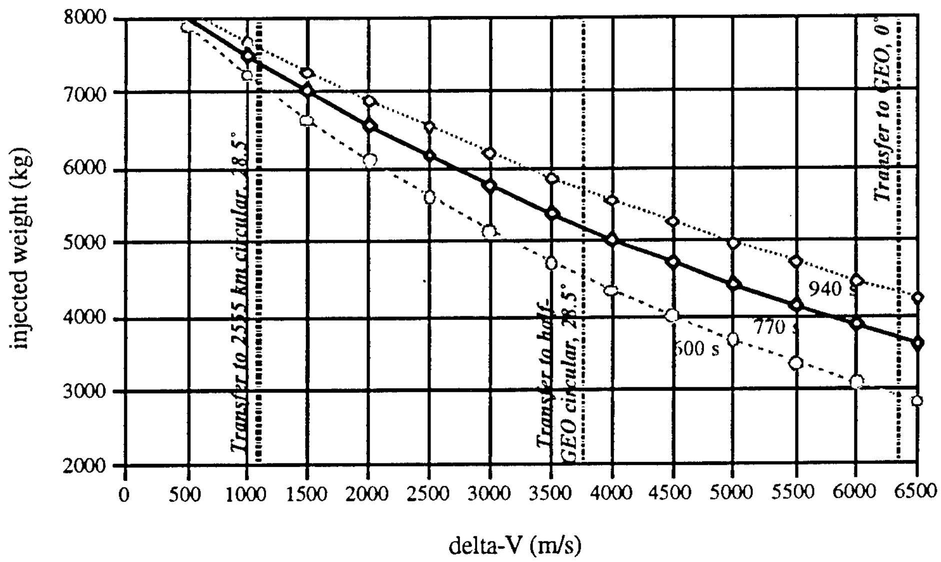


Figure 3.22 Assumptions for this chart include a launch from ETR into a 185-km circular parking orbit, a bimodal reactor system operating at a thrust-to-weight of 10^{-4}, and a variable I_{sp} (600, 770, and 940 s).

Figure 3.21 illustrates the capability of a bimodal satellite launched aboard a Delta II 7920 to a 185-km circular staging orbit. Given the three Isp values examined in the previous section, it was possible to determine the total injected weight the system could achieve. The Delta II 7920 can lift 3000 kg to the GATC altitude of 2555 km. A bimodal stage would increase this to between 4200 and 4500 kg. No values were tabulated for half-GEO, either in the JPL *Launch Vehicle Summary* or Ref. 11; the JPL study provides an elliptical orbit performance to this altitude of only 1700 kg. This mass would include that of the apogee kick motor required for circularization. The bimodal stage could place between 2600 and 3300 kg in this orbit.

By extrapolating from data in the *Atlas DoD Mission Planner's Guide* [Ref. 12] Atlas IIAS was determined to be capable of delivering an injected weight of 5600 kg to 2555 km. Substituting a bimodal system would increase this to 7000 kg or more (Fig. 3.22). At half-GEO, total delivery is down to a little over 2000 kg. A bimodal Atlas would more than double this.

Achievable injected weight for various delta-V requirements, Titan III booster

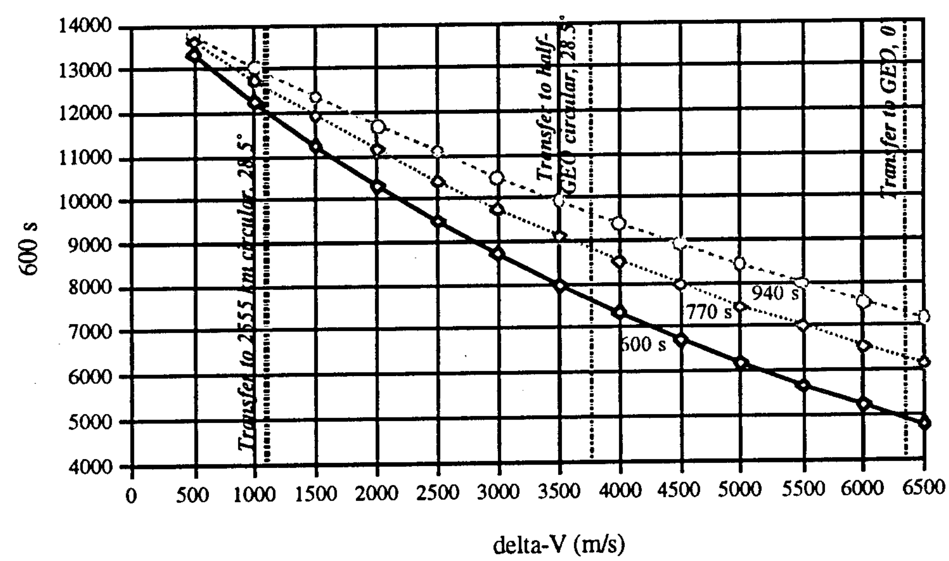


Figure 3.23 Assumptions for this chart include a launch from ETR into a 185-km circular parking orbit, a bimodal reactor system operating at a thrust-to-weight of 10^{-4}, and a variable I_{sp} (600, 770, and 940 s).

Figures 3.23 and 3.24 show bimodal satellite performance atop Titan III and Titan IV(SRMU) boosters. Neither booster can deliver any significant payload to 2555 km or half-GEO, unassisted; both require upper stages in order to reach these altitudes. A bimodal-augmented Titan IV is capable of placing between 19 and 20 metric tons to the

orbit used by the proposed GATC system. This performance is currently beyond the capability of any US launcher.

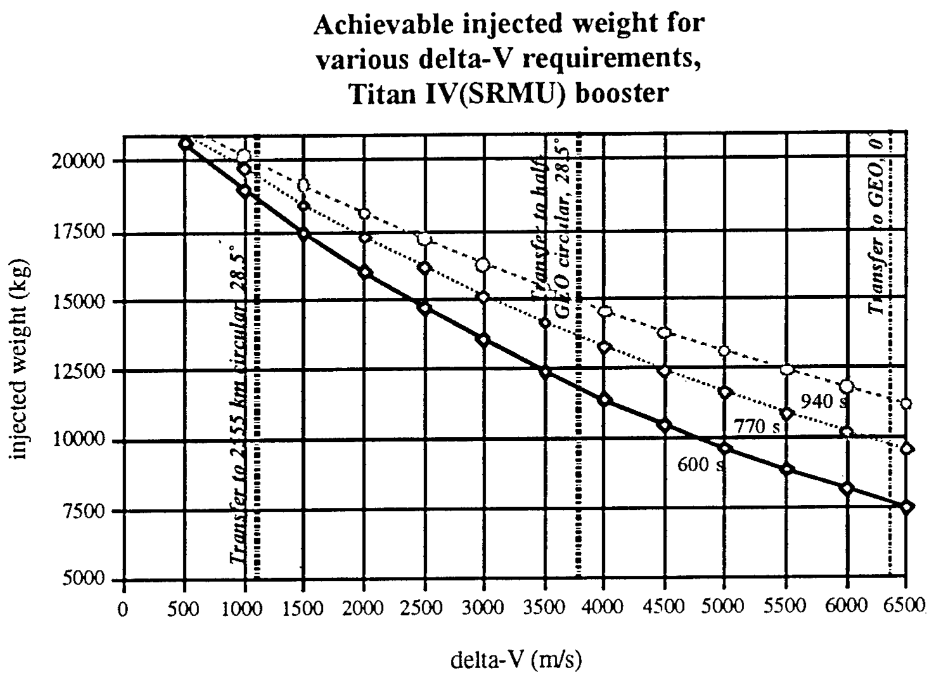


Figure 3.24 Assumptions for this chart include a launch from ETR into a 185-km circular parking orbit, a bimodal reactor system operating at a thrust-to-weight of 10^{-4}, and a variable I_{sp} (600, 770, and 940 s).

This capability supplements the very specific LEO-GEO orbital transfer analyzed in section 3.1. Even at the comparatively low altitudes considered--2555 km--a bimodal system would produce significant gains in payload delivery over current systems. Additionally, a bimodal satellite would enable Titan-class boosters to extend their operating envelopes without resorting to upper stage propulsion.

4.0 Utility Assessment

This section will examine how a bimodal system meets the requirements set forth in Section 3. Section 4.1 will cover the analysis that forms the basis for a bimodal system mass budget, while 4.2 will show how that budget affects the efficacy of a bimodal-powered satellite mission.

4.1 Bimodal System Mass Budget Estimation

The bimodal subsystem mass budget represents a significant portion of the total satellite mass. An upper mass limit can be estimated for three postulated bimodal systems for four classes of satellites, launched to LEO by a Delta II 7920, Atlas IIAS, Titan III/NUS, and Titan IV(SRMU)/NUS. For completeness, the following bimodal components are included in this analysis:

(1) the *reactor* and associated controls, shielding, and cooling
(2) *power conversion* , power conditioning and distribution
(3) *propulsion* system and control (both orbital transfer and stationkeeping /repositioning thrusters)
(4) *thermal management* (cooling loops, heat flow control, radiator, payload cooling capability, interfaces)
(5) *propellant management* (vessel, pumps, cryocooling, etc.)
(6) *guidance, navigation and control* set (including the communication link to ground control)

As a part of the utility assessment for bimodal satellite missions (section 4.2), it is useful to know that fraction of the mass required by equivalent subsystems of current satellites. The following approach to develop the mass budget is applied:

a) utilize available information on current satellite mass budgets,
b) acquire mass budget estimates for GN&C and propellant management systems from the available data,
c) derive the mass budget for the remaining subsystems,
d) perform a check to see whether the mass budget allocation is sufficient for the subsystems of a bimodal device, and
e) perform iterations on the mass budget allocations for the various classes of bimodal satellites, if the postulated limits are too low or too high.

4.1.1 Current Satellites

Table 4.1 [Ref. 7] lists the mass distribution of selected spacecraft by subsystem. Equivalent functions provided by the bimodal subsystem include the power, propulsion,

attitude determination and control (ADCS) subsystems, a portion of telemetry, tracking and command (TT&C), and a large portion of the thermal management subsystem (performing all of its own thermal management and supplying the payload heat sink - i.e. the radiator). Specialized payloads may require their own versions of these subsystems. In order to compare bimodal satellite capabilities with current satellites (e.g. delta-V budget and instrumented payload) we use the *average percentage of payload mass* shown as the last line of Table 4.1 to estimate the total mass of subsystems replaced on current technology satellite by a bimodal system. The bimodal subsystem functionally replaces the propulsion, power, and ADCS subsystems, for a total of 143% of the payload mass. The thermal management portion was not included in this summation since the majority of the current satellite thermal management task is handling the solar heat input and power dissipation within the payload and support subsystems. To obtain a more conservative estimate, the TT&C function was not included either (despite that a portion of TT&C function would be supplied by a bimodal device).

Percentage of Spacecraft Dry Mass by Subsystem

Satellite	Payload	Structure	Thermal	Power	TT&C	ADCS	Propulsion
FLTSATCOM 1-5	26.54	19.26	1.75	38.53	2.96	7.01	3.94
FLTSATCOM 6	26.38	18.66	1.99	39.39	2.99	6.77	3.83
FLTSATCOM 7-8	32.80	20.80	2.14	32.75	2.50	5.68	3.34
DSCS II	23.02	23.50	2.77	29.32	6.97	11.46	2.96
DSCS III	32.34	18.18	5.56	27.41	7.23	4.35	4.09
NATO III	22.12	19.29	6.51	34.74	7.51	6.33	2.43
INTELSAT IV	31.24	22.31	5.14	26.49	4.30	7.41	3.14
TDRSS	24.56	28.03	2.78	26.36	4.07	6.17	6.92
GPS Blk 1	20.49	19.85	8.70	35.77	5.84	6.16	3.61
GPS Blk 2, 1	20.15	25.13	9.86	30.97	5.20	5.41	3.29
GPS Blk 2, 2	23.02	25.37	11.03	29.44	3.10	5.25	2.68
P80-1	41.06	19.00	2.35	19.92	5.21	6.33	6.13
DSP 15	36.91	22.53	0.48	26.94	3.84	5.51	2.23
DMSP 5D-2	29.85	15.63	2.79	21.48	2.46	3.07	7.42
DMSP 5D-3	30.45	18.41	2.87	28.97	2.02	2.92	8.66
Average Percentage	**28.06**	**21.06**	**4.45**	**29.90**	**4.41**	**5.99**	**4.31**
Standard Deviation	**6.16**	**3.34**	**3.23**	**5.62**	**1.83**	**2.00**	**1.99**
Average % of Payload Mass	**100.00**	**75.06**	**15.86**	**106.55**	**15.73**	**21.34**	**15.36**

Table 4.1 - Mass Distribution for Selected Spacecraft

Table 4.2 shows the loaded, or *wet* weight of selected satellites used in Table 4.1, the approximate amount of propellant stored onboard, and the satellite dry mass (overall

platform mass without propellant). The available delta-V is derived from the amount of onboard propellant storage.

Spacecraft Name	Loaded Weight (kg)	Propellant Mass (kg)	Dry Mass (kg)	delta-V (m/s); Isp=240 s, 310s
FLTSATCOM 1-5	930.9	81.4	849.5	215.4 278.3
FLTSATCOM 6	980.0	109.1	870.9	277.9 358.9
FLTSATCOM 7-8	1150.9	109.0	1041.9	234.3 302.6
DSCS II	530.0	54.1	475.9	253.5 327.4
DSCS III	1095.9	228.6	867.3	550.8 711.5
NATO III	346.1	25.6	320.4	181.7 234.6
INTELSAT IV	669.2	136.4	532.8	536.7 693.2
TDRSS	2150.9	585.3	1565.7	747.6 965.7
GPS Blk 1	508.6	29.5	479.1	140.7 181.7
GPS Blk 2, 1	741.4	42.3	699.1	138.3 178.7
GPS Blk 2, 2	918.6	60.6	858.0	160.7 207.5
P80-1	1740.9	36.6	1704.4	49.9 64.4
DSP 15	2277.3	162.4	2114.9	174.2 225.0
DMSP 5D-2	833.6	19.1	814.6	54.3 70.1
DMSP 5D-3	1045.5	33.1	1012.3	76.0 98.0

Table 4.2 Loaded Weight, Propellant Mass, and Dry Mass of Selected Satellites
(Available delta-V is calculated for I_{sp}=240 s and 310 s, corresponding to monopropellant and bipropellant hydrazine thrusters, respectively.)

4.1.2 GN&C set

A fully autonomous guidance, navigation and control unit would use the global positioning system (GPS) satellites to provide parameters for exact orbit location, It will have an attitude determination and control set (ADCS), an inertial measurement unit (IMU) based on laser gyros and low-g accelerometers, a star sextant, dedicated computer hardware and software, relevant sensors, and support structure. The listed technology data (Table 4.3) are from Ref.'s 7 and 13; the total GN&C set mass is estimated to be less than 100 kg.

Component	Manufacturer	Function	Size (cm3)	Mass (kg)	Power (W)
GPS (2 units total)	Honeywell	Orbit location determination	4,000 x 2	4 x 2	35 x 2
or GPS (2 units total)	Rockwell	" "	4,700 x 2	4 x 2	12 x 2
Space Sextant	Martin Marietta	Orbit & Attitude	4×10^5	25	50
Inertial Measurement Units (2) - IMU	Honeywell	local inertial reference, GPS updated	est. 10,000	est.5 x 2	est.100
Computers (2-3 units total)	Custom	control GN&C systems	est. 10,000	est.10	est.200
Telemetry/Communication	" "	some info to ground control	est. 5,000	est. 5	est.100
Other components			<10,000 volume reserve	<40 mass reserve	<50 power reserve
TOTAL			**est. < $5 \times 10^5 cm^3$**	**est. < 100 kg**	**est. < 600 W**

Table 4.3 - Size, mass, power requirements of autonomous GN&C hardware elements

4.1.3 Propellant Management

The propellant (liquid H_2, abbreviated LH_2) is contained in a vessel equipped with all necessary refrigeration and pumping equipment. To derive the dry vessel mass, a tankage fraction of 15% is assumed.

As an example, consider a bimodal communication satellite lifted to 185 km by Atlas IIAS; the bimodal system subsequently is used to transfer the satellite to GEO. The maximum initial mass at LEO is 8560 kg (refer to Fig. 3.2). For a postulated 1500 kg bimodal device, there is a total of 7060 kg available for the propellant, payload, and those subsystems which are not replaced functionally by the bimodal system.

From Fig.'s 3.2 and 3.11, it is seen that the propellant mass needed to get to GEO from 185 km is approximately 4900 kg. The amount of propellant spent to reach the destination orbit depends only on (1) the initial mass in low earth orbit, (2) the bimodal system thrust-to-weight, and (3) the system specific impulse. It is entirely independent of the satellite subsystem mass breakdown. The required delta-V for a low-thrust transfer from 185 km to GEO is 6300 m/s (assuming sequential firings for circularization and plane changes). Using the tankage fraction above, the propellant vessel mass is 735 kg. For purposes of this report, it is assumed that the orbital transfer propellant and the on-orbit maneuvering propellant are contained in separate tanks; the orbital transfer tank is jettisoned following insertion at GEO. An additional 131 kg are allocated for (1) cryogenic hydrogen--or perhaps a storable propellant for electric thrusters, which would

be more efficient--for GEO maneuvering and stationkeeping (114 kg), and (2) the on-orbit maneuvering tank (17 kg, or 15% of the total propellant mass contained). Similar analyses can be performed for a heavier bimodal satellite launched by Titan IV (SRMU) / NUS (vessel mass = 1913 kg) or for a lighter satellite launched by the Delta II 7920 (vessel mass = 420 kg).

Current systems' delta-V budgets account mainly for stationkeeping, which requires approximately 50 m/s per year of satellite operation; a ten-year platform would thus have a delta-V budget of 500 m/s. For the bimodal satellite launched from Delta II, the 114 kg of LH_2 in the on-orbit maneuver tank would provide this velocity change capability.

4.1.4 Instrumented Payload Concept

The bimodal unit replaces several subsystems of current technology satellites. The *instrumented payload* is defined as the total dry mass of those subsystems which are *not functionally replaced* by the bimodal device.

For the utility assessment analysis (section 4.2), which compares bimodal and current satellites, the maximum instrumented payload of four contemporary booster/upper stage combinations is derived (Delta II 7920/PAM-D, Atlas IIAS, Titan III, and Titan IV(SRMU)/ Centaur). The data for Titan III is not presented in this section, although the system was examined in a similar manner.

The maximum injected weight delivered by Atlas IIAS to GEO is 1050 kg (Fig. 3.2). From Table 4.2, it can be seen that a typical current satellite has about 10% of its loaded mass reserved for the propellant. Therefore, the current maximum dry mass is about 945 kg. Table 4.1 demonstrates that the average total mass of subsystems which are not functionally replaced by the bimodal subsystem is 58% of the dry mass budget, which in this case is close to 550 kg . Therefore, Atlas IIAS has an instrumented payload (IP) delivery capability to GEO of approximately 550 kg. This can also be derived for other vehicles, yielding values of 2700 kg for Titan IV(SRMU)/Centaur and 475 kg for Delta II 7920/PAM-D. Although not shown, Titan III/TOS provides 710 kg of instrumented payload and a loaded IP of 846 kg.

Launcher and maximum injected weight at GEO	Delta II 7920/PAM-D **910 kg**	Atlas IIAS **1050 kg**	Titan IV(SRMU)/Centaur **5200 kg**
Instrumented payload	475 kg	550 kg	2700 kg
Loaded instrumented payload	566 kg	655 kg	3220 kg
Estimated Max. Power	2.73 kW_e	3.15 kW_e	15.60 kW_e

Table 4.4 Current Instrumented Mass Delivery Capabilities to GEO

The *loaded*, or *wet* instrumented payload is obtained by adding the propellant weight, initially subtracted from the injected weight, to the instrumented payload. *Estimated maximum power* is derived by assuming a total power system mass of approximately 30% of the injected weight. An advanced solar power system with a specific mass of 100 kg/kW_e is used to determine the total power output, based on the calculated power system mass. These values are shown in Table 4.4.

Obviously, there are variations from these derived masses for specific missions; however, this approach appears valid for the comparison of respective capabilities at various orbits.

4.1.5 Bimodal Mass Budget -- Parametrics

In this section, a range of bimodal system masses are examined. A fairly significant portion of the bimodal system is given over to the reactor fuel, moderator, reflector, control systems, shielding, power conversion, propulsion, and the thermal management system (e.g. radiator). However, the actual masses of these items are exceedingly concept-dependent and it is not the intention of this study to select a specific reactor scheme or set of schemes. Therefore, three mass levels were studied: *600*, *1100*, and *1600* kg. This includes the aforementioned parts of a *bimodal reactor system* (BRS) but excludes the propellant management and GN&C budgets. As done previously, an I_{sp} of 770 s and a thrust-to-weight ratio of 10^{-4} is assumed. For all three launchers to LEO we have considered, the bimodal satellite mass delivered to GEO is significantly higher than the mass of current platforms. Total power available to these platforms also varies by concept, but it is noted that two reactor designs (Rocketdyne's *S-PRIME* and SPI's *SPACE-R*) currently being studied would provide *40* kW_e and weigh approximately 2100 kg [Ref.'s 14, 15]. INEL's *SEHPTR* would provide the same power at a weight of 2600 kg [Ref. 16].

Results of this parametric analysis can be seen in Table 4.5. The key figure of merit for comparison is the bimodal satellite loaded instrumented mass. Note that, in the case of Delta II, a bimodal unit will break even with a conventional one if its reactor system (reactor, shielding, power conversion, etc.) weighs less than 1100 kg. There is greater margin available with the larger boosters; the breakeven reactor system mass for Atlas IIAS, ~ 2100 kg, can be extrapolated from the table. For Titan IV, this rises to almost 4500 kg.

	Delta II 7920			Atlas IIAS			Titan IV(SRMU)/NUS		
propellant tankage (on-orbit maneuver)	*17*	*17*	*17*	*29*	*29*	*29*	*76*	*76*	*76*
propellant tankage (orbital transfer)	*420*	*420*	*420*	*735*	*735*	*735*	*1913*	*1913*	*1913*
bimodal reactor system	*600*	*1100*	*1600*	*600*	*1100*	*1600*	*600*	*1100*	*1600*
GN&C system	*100*	*100*	*100*	*100*	*100*	*100*	*100*	*100*	*100*
GEO injected weight	2200	2200	2200	3750	3750	3750	9750	9750	9750
bimodal satellite loaded instrumented payload mass	***1063***	***563***	***63***	***2286***	***1786***	***1286***	***7067***	***6567***	***6067***
current satellite loaded instrumented payload mass	566	566	566	655	655	655	3220	3220	3220
IMLEO (185 km)	5000	5000	5000	8560	8560	8560	22500	22500	22500

Table 4.5 Bimodal Satellite Instrumented Masses for Various Booster Systems
(Assumptions for the bimodal satellite include a thrust-to-weight ratio of 10^{-4} and a specific impulse of 770 s; propellant tankage masses are derived by assuming their weight is equivalent to 15% of their contained propellant mass; a total on-orbit maneuver delta-V of 500 m/s is used. All masses are in kilograms.)

	Delta II 7920			Atlas IIAS			Titan IV(SRMU)/NUS		
reactor system mass	600 kg	1100 kg	1600 kg	600 kg	1100 kg	1600 kg	600 kg	1100 kg	1600 kg
bimodal satellite instrumented payload mass (770 s)	***949***	***449***	--	***2092***	***1592***	***1092***	***6565***	***6065***	***5565***
(600 s)	***443***	--	--	***1262***	***762***	***262***	***4266***	***3766***	***3266***
(940 s)	***1348***	***848***	***348***	***2819***	***2319***	***1819***	***8011***	***7511***	***7011***
current satellite instrumented payload mass	475			550			2700		

Table 4.6 Bimodal Satellite Instrumented Masses for Varying Specific Impulse and Bimodal Reactor System Mass
(Assumptions for the bimodal satellite include a thrust-to-weight ratio of 10^{-4}. Propellant tankage masses are derived according to the algorithm mentioned in Table 4.5. All masses are in kilograms, and are 'dry': this is not loaded mass. Shaded boxes indicate system combinations which do not exceed the current booster/satellite capability.)

Similar values can be generated for 600 and 940 s I_{sp} systems. Table 4.6 illustrates the instrumented payload mass available to the three launch systems analyzed. (The 'dry'

mass is used here to compare with the selection of charts in section 4.2, which plot dry mass versus available maneuver delta-V in GEO.) At a bimodal system specific impulse of 600 s, all reactor systems are too large for the bimodal satellite to meet Delta's current capability. Even at 770 s, the reactor system would have to weigh less than 1100 kg to make it competitive with the current Delta booster.

For the case of satellite platforms launched off of Atlas IIAS and Titan IV(SRMU)/NUS, the bimodal satellite loaded mass exceeds the capability of the conventional launcher in every case but one. In the case of Titan, the 770 s and 940 s systems at least *double* the instrumented payload mass to GEO.

4.2 Utility Assessment

In the previous section, the increase in GEO payload mass made available by bimodal satellite technology was determined. In order to provide a fair comparison with current capabilities, the concept of *instrumented payload* was developed in section 4.1; by including a parametric analysis of the weight of the bimodal system itself, it is possible to determine the true benefit derived from using a power and propulsion concept. Without this analysis, the propulsion trades are reduced to comparisons of current and predicted specific impulse and thrust-to-weight ratios and their effect on payload. This can be misleading since it is not immediately apparent how massive a bimodal power and propulsion will be; there are a number of plausible designs, some weighing as much as 4000 kg. It is possible that, due to high weight, a bimodal system will not be able to achieve performance gains over conventional systems. This section will provide weight goals for such a system, based on supposed requirements for booster stepdown and maneuverability.

It is necessary to briefly revisit the instrumented payload concept--the average propellant mass of a current geosynchronous satellite platform is about 10% of its loaded or wet weight. Therefore, the average dry mass of these systems is about 90% of the loaded mass. From the dry mass budget we determine the instrumented payload by subtracting the mass of each subsystem replaced by the bimodal subsystem.

The maximum loaded instrumented payload (MLIP) in GEO for current satellites is obtained by adding the propellant mass to the instrumented payload shown above. The figure derived for MLIP can be compared with a similar figure for bimodal satellites to determine enhancements in the payload and delta-V budgets. The delta-V budget available to a bimodal satellite can therefore be compared to today's capabilities.

4.2.1 Atlas IIAS Utility

Figure 4.1 shows the operating envelopes of bimodal satellites at GEO in terms of available delta-V and instrumented mass. The delta-V budgets for conventional systems was derived by assuming a 10% propellant mass fraction (propellant weight / injected weight); for a 280 s bipropellant hydrazine system, this translates into a delta-V of approximately 300 m/s.

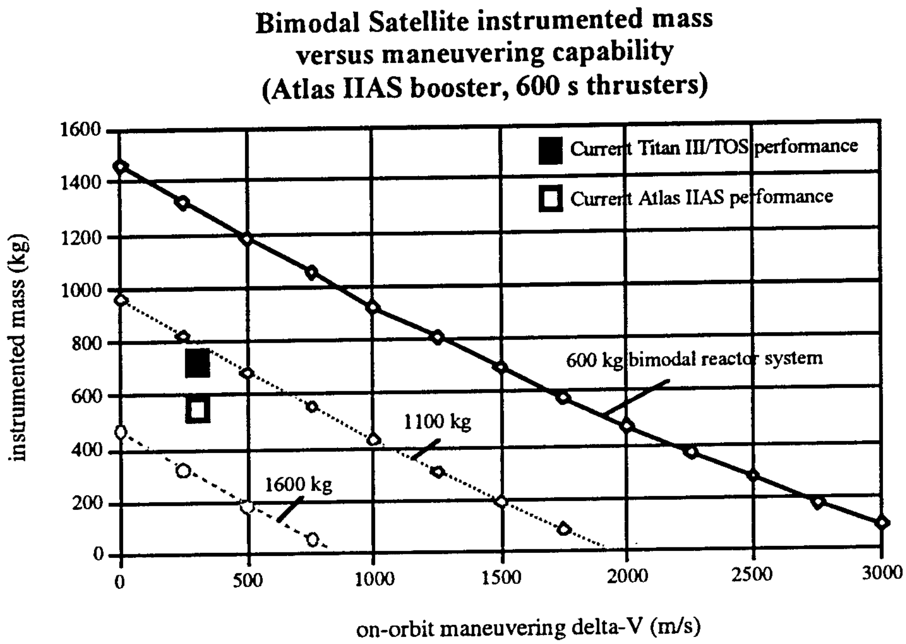


Figure 4.1 The performance of an Atlas IIAS booster with a bimodal system (total reactor system mass of 600, 1100, and 1600 kg) performing a continuous burn from LEO to GEO is compared to conventional booster/upper stage combinations. The altitude of the starting orbit is 185 km. The bimodal system achieves a thrust-to-weight ratio of 10^{-4} and a specific impulse of 600 s (H_2 exhaust temperature of approximately 1215 K).

A bimodal reactor system at this specific impulse only becomes competitive when its mass falls below approximately *1350 kg*; the current Atlas IIAS places 550 kg of instrumented mass in GEO with a delta-V budget of 300 m/s, falling between the 1100 and 1600 kg bimodal lines. Below 1100 kg, the system also outperforms Titan III/TOS. Note that an 1100-kg BRS has approximately twice the delta-V capability of a current system; if a 600-kg is achievable, the total delta-V budget may rise to almost 2300 m/s, *seven times* the current capability. Recall that a 30°/day repositioning maneuver in GEO needed a maximum of 200 m/s.

Similarly, the bimodal system enables increased instrumented mass on-orbit. For a fixed maneuver budget of 300 m/s, a 600-kg BRS provides an instrumented mass of almost 1300 kg, a 230% increase over the current Atlas IIAS.

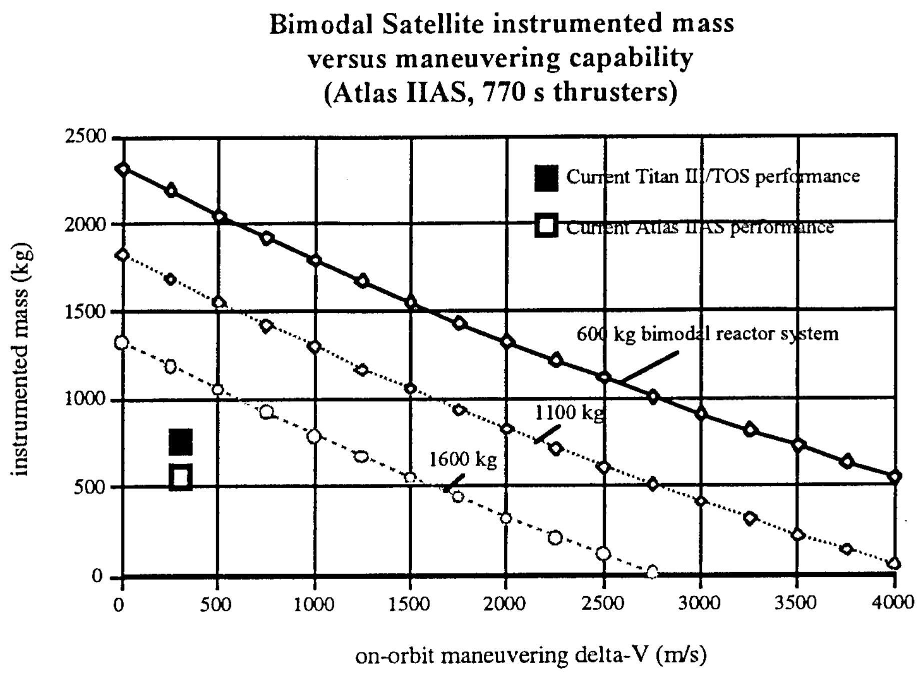


Figure 4.2 Assumptions are the same as in the previous figure. The depicted bimodal system achieves a specific impulse of 770 s (H_2 exhaust temperature of approximately 2000 K).

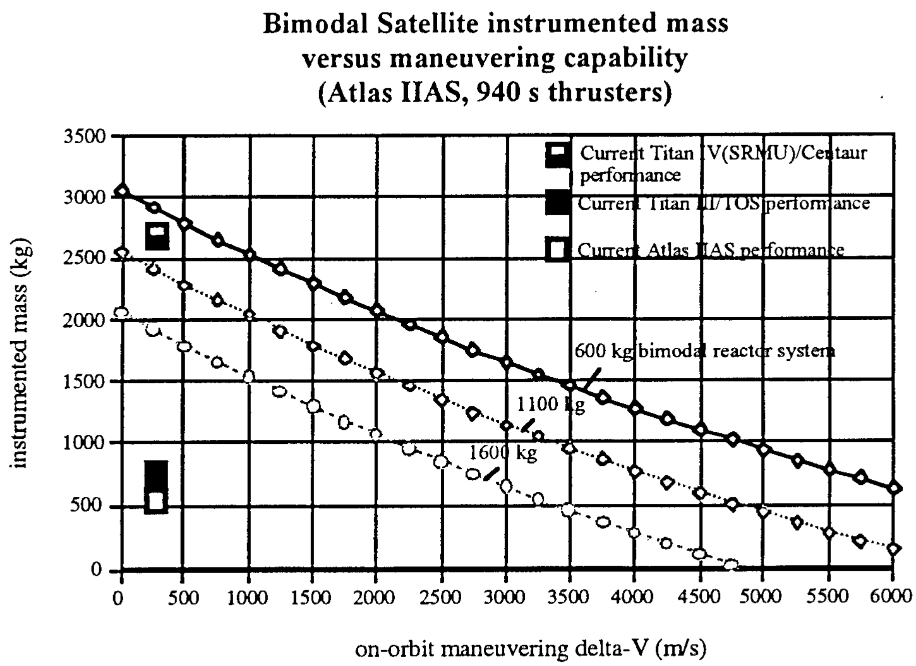


Figure 4.3 Assumptions for this chart are the same as those in Figures 4.1 and 4.2. The bimodal system achieves a specific impulse of 940 s (H_2 exhaust temperature of approximately 3000 K).

The increase in I_{sp} from 600 to 770 s (Fig. 4.2) allows even the heaviest bimodal reactor system to exceed the capabilities of the conventional Atlas IIAS and Titan III/TOS. At

600 kg, a BRS permits either (1) the current Atlas IIAS payload and a maneuver budget of 4000 m/s, or (2) a 2200-kg instrumented payload with the current 300 m/s maneuver budget.

By increasing specific impulse to 940 s, it is possible to achieve stepdown of Titan IV(SRMU)/Centaur payloads to a bimodal-augmented Atlas IIAS (Fig. 4.3). However, the bimodal reactor system must weigh 600 kg. These results match those of section 3.1, when it was determined that only a high-thrust, high-Isp BRS could permit stepdown from Titan IV to Atlas.

Interestingly, the 940 s BRS achieves a delta-V capability of more than 6000 m/s for the nominal Atlas IIAS 550-kg instrumented payload. This is nearly sufficient to return the system to low earth orbit, if necessary.

4.2.2 Titan III Utility

As explained earlier, the analysis for Titan III was not presented in section 4.1. However, the smaller Titan system is a very capable platform for a bimodal satellite, as will be shown.

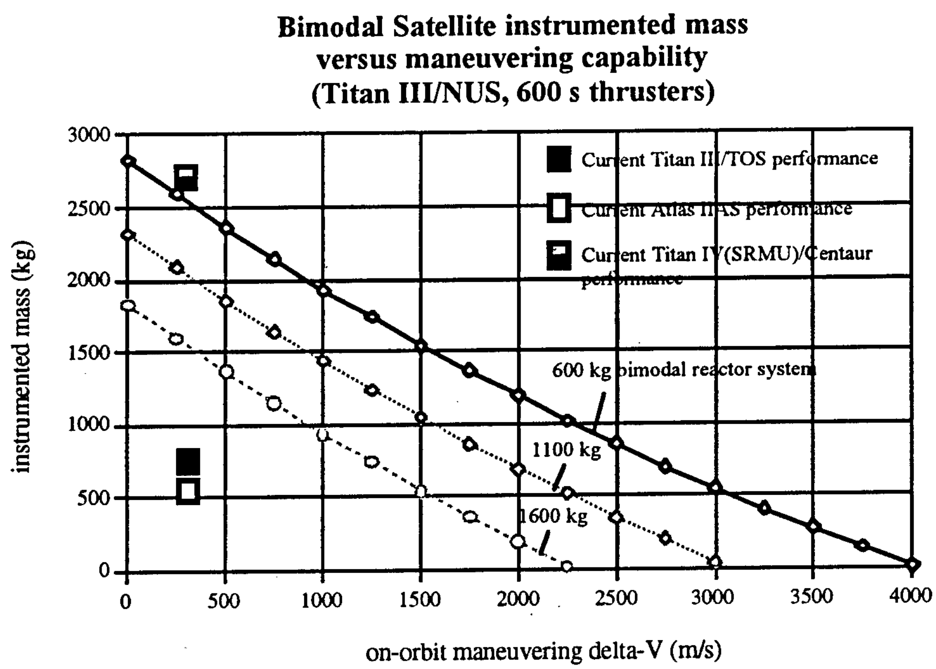


Figure 4.4 This figure is similar to the three Atlas IIAS charts in the previous section. The performance of a Titan III booster with a bimodal system (total reactor system mass of 600, 1100, and 1600 kg) performing a continuous burn from LEO to GEO is compared to conventional booster/upper stage combinations. The bimodal system achieves a specific impulse of 600 s (H_2 exhaust temperature of approximately 1215 K).

The operating envelope of a Titan III with a 600 s bimodal satellite is illustrated in Figure 4.4. At 600 kg, the bimodal reactor system's performance is nearly sufficient for it to meet the stepdown requirement (from Titan IV(SRMU)/Centaur).

In Figure 4.5 below, it is clear that even the most massive system considered (1600 kg) still allows the bimodal Titan III to achieve greater instrumented payloads than Titan IV(SRMU)/Centaur. With this largest BRS, the delta-V achieved is approximately 750 m/s, twice that of the conventional system shown.

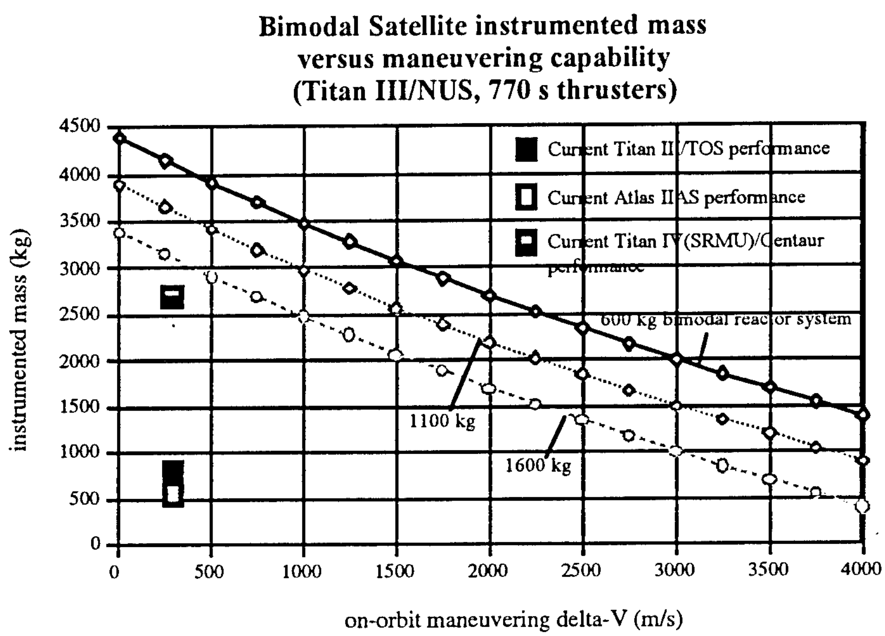


Figure 4.5 Titan III/NUS achieves stepdown for even the heaviest bimodal reactor system shown (1600 kg); a bimodal system could weigh as much as 2000 kg and still meet this stepdown. The BRS achieves a specific impulse of 770 s (H_2 exhaust temperature of approximately 2000 K).

Since the 770 s BRS meets the stepdown requirement for all three masses examined, the 940 s bimodal reactor system is not shown.

4.2.3 Titan IV(SRMU)/NUS Utility

As of this writing, Titan IV is the largest expendable launch vehicle in the US inventory; it is capable of boosting 5200 kg to GEO and is theoretically capable of over 5700 kg, given a structural redesign. The Russian *Energia* booster can place up to 18000 kg in geosynchronous orbit. As demonstrated in section 3.0, a stepdown of an *Energia*-class payload to Titan IV(SRMU)/NUS is not possible with the bimodal systems examined.

Bimodal Satellite instrumented mass
versus maneuvering capability
(Titan IV(SRMU)/NUS, 600 s thrusters)

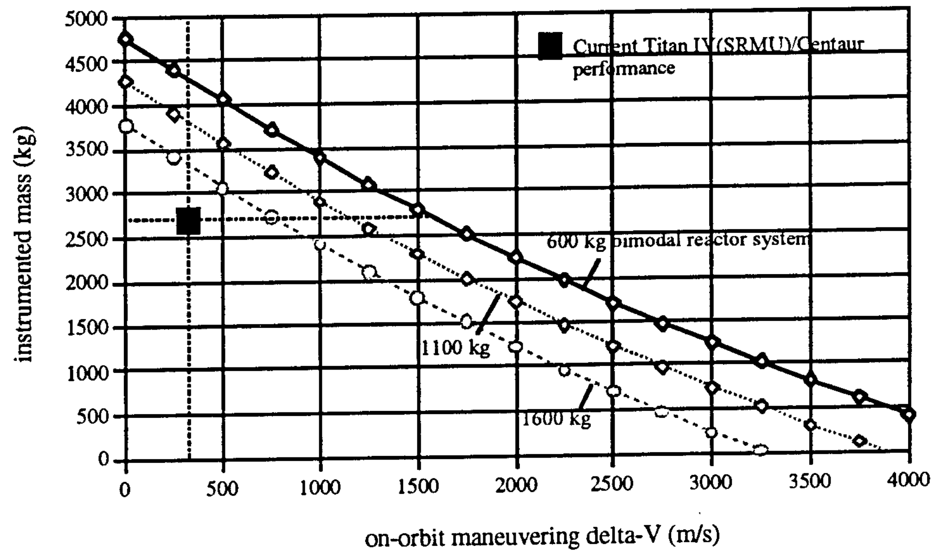


Figure 4.6 This chart displays the performance envelope of a bimodal satellite launched from a Titan IV(SRMU)/NUS and compares it to the current estimated capability of Titan IV(SRMU)/Centaur. The bimodal system achieves a specific impulse of 600 s (H_2 exhaust temperature of approximately 1215 K). Dotted lines are provided to determine (1) increased maneuver capability for a fixed mass, and (2) increased instrumented payload for a fixed delta-V budget.

Bimodal Satellite instrumented mass
versus maneuvering capability
(Titan IV(SRMU)/NUS, 770 s thrusters)

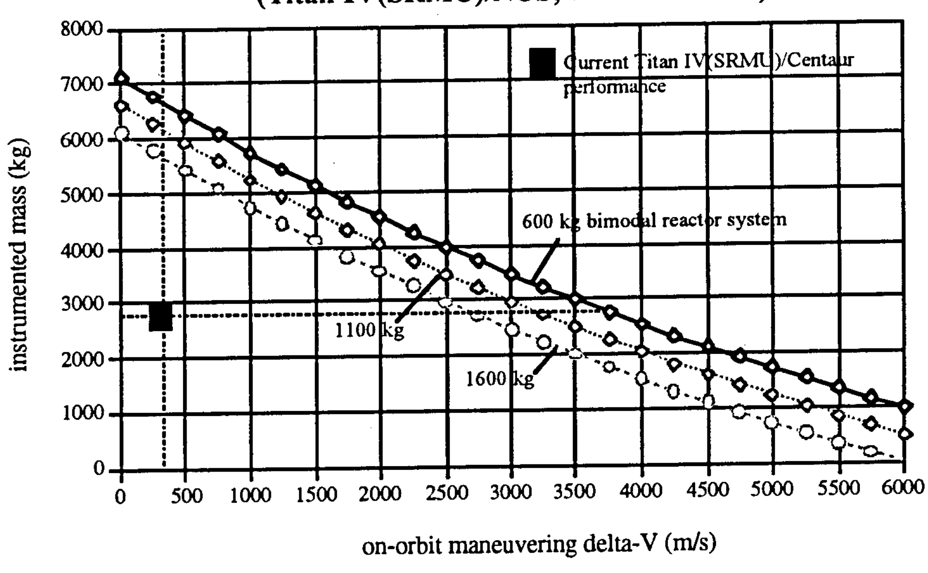


Figure 4.7 This chart shows the performance of Titan IV(SRMU)/NUS with a bimodal satellite (as in Fig. 4.6). The bimodal system achieves a specific impulse of 770 s (H_2 exhaust temperature of approximately 2000 K).

At 600 s, all bimodal systems exceed the current Titan IV's performance. This is substantially different from the results for Delta (seen in the following section) and Atlas IIAS. Neither of these smaller launchers provides enough mass margin for the heavier bimodal systems to achieve performance gains at the low specific impulse of 600 s.

At 770 s, the BRS allows Titan IV(SRMU)/NUS to double its instrumented payload delivery (>6500 kg). For a fixed instrumented payload mass of 2700 kg, the largest bimodal system examined allows a large increase in the delta-V budget, from 300 to 2750 m/s.

Once again, the 940 s system is not shown.

4.2.4 Delta II 7920 Utility

The Delta II class booster was the smallest system examined in this study. Delta/PAM-D provides an injected weight at GEO of 910 kg and a calculated instrumented mass of 475 kg. For purposes of the following utility assessment, a Delta II 7920 was assumed to lift 5000 kg to a 185-km circular orbit; a bimodal satellite would separate from the booster and perform a continuous burn from LEO to GEO.

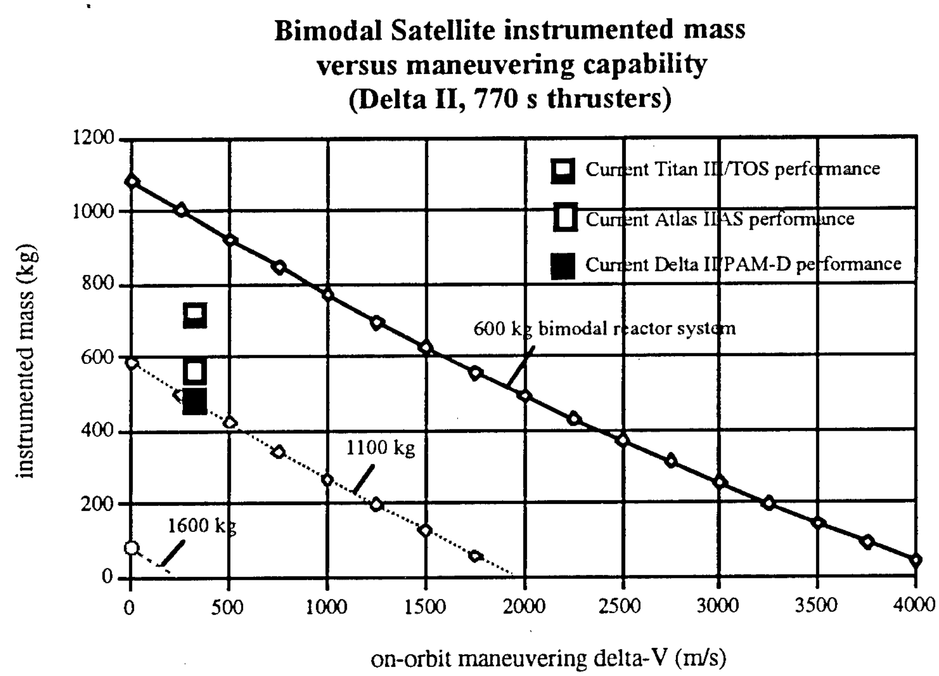


Figure 4.8 The performance of a Delta II 7920 booster with a bimodal system (total reactor system mass of 600, 1100, and 1600 kg) performing a continuous burn from LEO to GEO is compared to conventional booster/upper stage combinations. The altitude of the starting orbit is 185 km. The bimodal system achieves a specific impulse of 770 s (H_2 exhaust temperature of approximately 2000 K).

A 600 s bimodal reactor system was incapable of exceeding the performance of the current Delta II 7925/PAM-D even at the lowest BRS mass of 600 kg. The performance

envelope for the 600 s system is not shown. Figure 4.8 shows that, at a specific impulse of 770 s, the BRS must (1) weigh less than 1100 kg to break even with its conventional competitor and (2) weigh less than ~850 kg in order for the bimodal satellite to enable a stepdown from Atlas IIAS to Delta II.

Bimodal Satellite instrumented mass versus maneuvering capability (Delta II 7920, 940 s thrusters)

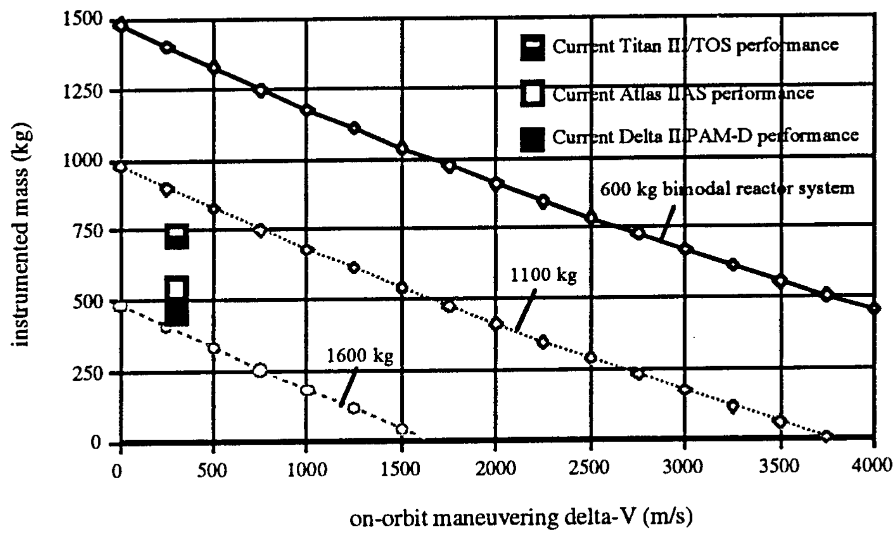


Figure 4.9 Assumptions are the same as in the previous chart. The bimodal system achieves a specific impulse of 940 s (H_2 exhaust temperature of approximately 3000 K).

Increasing the BRS specific impulse to 940 s will allow it to achieve an Atlas-to-Delta stepdown at a higher system weight (~1250 kg). Titan III-to-Delta stepdowns are also possible.

Stepdown Type	*Launch Cost Savings (FY92$)*	*Bimodal Reactor System Requirements*
Titan IV/Centaur to Titan III/Bimodal	**$255M**	>1600 kg at 940 s >1600 kg at 770 s ≤600 kg at 600 s
Titan IV/Centaur to Atlas IIAS/Bimodal	**$280M**	≤600 kg at 940 s Cannot be achieved at 600 or 770 s
Atlas IIAS to Delta II 7920/Bimodal	**$70M**	≤850 kg at 770 s ≤1250 kg at 940 s

Table 4.7 Bimodal Reactor System Requirements for Launch Vehicle Stepdown
Three mass levels were examined: 600, 1100, and 1600 kg bimodal systems. These masses do not include the weight of propellant tanks, propellant, or the guidance subsystem. (The cost figures come from the JPL Launch Vehicles Summary for Mission Planning, JPL D-6936 Rev. C. This layout follows the format of Table 3.3.)

Table 4.7 shows the bimodal reactor system mass requirements needed to achieve the stepdowns described in section 3.0. Comparing these values with estimated mass budgets for other power and propulsion reactors provides a clue as to the amount of developmental effort that will be needed to produce a working bimodal system.

Titan IV payloads can be moved to either Titan III or Atlas IIAS. At 600 s, a stepdown to Titan III will require a very compact reactor system, with a total mass somewhat under 600 kg. The move to Atlas IIAS is not possible except at a specific impulse of 940 s. This agrees with the results of section 3.1, which showed that Titan IV payloads were capable of being placed aboard an Atlas if the bimodal system could achieve high thrust-to-weight and high specific impulse.

Atlas IIAS and Titan III/TOS payloads can be moved to Delta launchers but only at the highest two I_{sp}'s examined. There is a fairly stringent weight requirement on the system in both cases.

5.0 Summary and Conclusions

A number of parametric trades were performed to determine the level of performance a bimodal power and propulsion system would have to achieve to meet the following broad objectives: (1) reduce launch and satellite operational costs, (2) enhance existing mission capabilities by providing greater payload or maneuverability, and (3) enable new missions that require high power, high weight or maneuverability. These trades led to the following bimodal reactor system specifications (Final Operational Capability):

Bimodal Reactor System Mass — ***850 kg (1870 lbs)***
2000 kg (4400 lbs)

(Coupled to the specific impulse requirement, the more stringent requirement permits Titan IV/Centaur to Titan III, Titan III/TOS to Delta, and Atlas IIAS to Delta stepdowns; this figure includes reactor fuel, moderator, reflector, control systems, shielding, power conversion, thermal propulsion, and the thermal management system. The less stringent--2000 kg--requirement would permit only the Titan IV/Centaur to Titan III stepdown.)

Thermal propulsion system thrust — ***80 N (18.2 lbs)***
(allows 1-week transfer of 5200-kg injected weight to GEO; this is the amount of mass that a Titan IV(SRMU)/Centaur can deliver to GEO)

Thermal propulsion system specific impulse — ***770 s***
(implies H_2 exhaust temperature of 2000 K)

Electrical Power Output — ***8.5 kW$_e$***
20 kW$_e$

(Current maximum power output for Delta II GEO payloads was derived in section 4.1.4 for a specific mass of 100 kg/kW$_e$. The two figures shown here assume the same specific mass, given the bimodal reactor system masses required for a bimodal Delta II 7920 and Titan III/NUS.)

Operational lifetime (power production) — ***10 years***
(MILSTAR requires 10-year operation)

Thermal propulsion system operating time — ***250 hours***
(assumes one week of continuous thrusting for orbital transfer plus a 50% margin for on-orbit maneuvering)

While the results of section 3.0 confirm that electric propulsion devices appear optimal for low-thrust maneuvers (such as GEO repositioning), no recommendations on their specifications have been made. This is because, unlike the thermal propulsion system, the electric thrusters will not be integral to the bimodal reactor system itself.

These specifications were driven by the analysis in sections 3.0 and 4.0. Section 3.1 focused on the concept of launch vehicle stepdown: Moving payloads from larger boosters (e.g. Titan IV) to smaller, less expensive vehicles such as Atlas IIAS and Delta. This is the most dramatic method of achieving the first objective, with potential launch cost savings as high as \$280M/booster. It was determined that bimodal power and propulsion systems could have specific impulses as low as 600 s (using hydrogen, equivalent to an exhaust temperature of just over 1200 K) and still move a geosynchronous payload from a Titan IV/Centaur to a Titan III/NUS. It was also possible to move Atlas IIAS GEO payloads to Delta boosters. This is significant because thermoelectric power conversion systems would run at similar temperatures and are within the capabilities of current technology.

In order to move Titan IV/Centaur payloads onto the Atlas IIAS, the bimodal satellite must be capable of 940 s of Isp and vehicle thrust-to-weight ratios approaching 1.0. This is beyond the performance achieved by any current system, including the NERVA rockets of the 1960's. Additionally, this level of performance made possible the placement of very large satellites (as much as 12,250 kg) in GEO. The bimodal system's combined high-performance propulsion and high-power (tens of kilowatts) *enable* such heavy satellite platforms as the proposed Direct Broadcast High Definition Television Satellite. A bimodal system requirements list was devised, containing the following: (1) a specific impulse of 770 s, and (2) a thrust level of ~ 80 N in order to achieve 7-day LEO-GEO transfer of a MILSTAR-class satellite (this could be relaxed to 20 N if the transfer time could be extended to 28 days).

The level of performance achieved by the bimodal system was extremely sensitive to the minimum starting altitude. This is not a function of the bimodal system itself; it was shown in section 3.1 that the difference in delta-V requirements between a 185-km start and a 1000-km start is approximately 400 m/s. The factor that most influences performance is the throwweight provided by the various boosters to the start altitude. Atlas and Delta are the most forgiving boosters in that their payload delivery falls off by only 20% over the range of interest. In contrast, systems like Titan III and Titan IV lose 90% or more of their payload delivery capability as the destination orbit is raised from 185 to 1000 km. This severely limits the quantity of mass that the bimodal system can place in geosynchronous (or any final) orbit.

Two maneuvering enhancements were analyzed in detail: GEO repositioning and sun-synchronous orbit repositioning. Military systems make use of both these orbits but are not generally capable of making large changes during their operating lifetimes. A bimodal system could achieve a longitudinal drift rate in GEO of up to 30°/day at one-third to one-

sixth the mass cost of conventional hydrazine systems used today. These same propellant savings would be realized for repositioning satellites in sun-synchronous orbits.

A detailed study of the mass budgets of current satellite systems led to trade studies involving variable-mass bimodal reactor systems. It was shown that Titan IV/Centaur payloads could be stepped down to Titan III/NUS for the largest (1600 kg) bimodal reactor system if the unit were capable of performing at I_{sp}'s above 770 s. At 600 s, this stepdown could be achieved but only if the reactor system could be built within a budget of 600 kg. Other stepdowns (Titan IV/Centaur to Atlas IIAS, Atlas IIAS to Delta) also required small reactor systems--less than 1250 kg.

As shown, bimodal power and propulsion would significantly lower launch costs by allowing launcher stepdown. The additional payload that allows stepdown can be used aboard the larger boosters (such as Titan IV) to enable the deployment of large systems such as the Global Air Traffic Control satellite; these systems cannot be placed in their operational orbits by existing US boosters. Furthermore, their power requirements are high (10-100 kW_e) and are not easily met by solar photovoltaic systems. Additionally, a bimodal system would provide much more efficient on-orbit maneuvering than is currently available, cutting propellant use by as much as a factor of seven.

Acknowledgments

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Appendix A Specific Impulse Issues

Specific impulse should be maximized, within the constraints placed by reactor design and lifetime considerations. Material properties limit propellant temperatures to around 3500K. The Particle Bed Reactor (PBR) has the potential for achieving this level of performance, but this has yet to be demonstrated.

The bimodal system's capability will be determined by its reactor operating temperature; the choice of nuclear fuel will place an upper limit on operating life at a given temperature, while the choice of propellant will determine the achievable specific impulse and therefore the bimodal satellite payload delivery capability. Cryogenic hydrogen (LH_2) offers the best performance of any propellant, due its extremely low molecular weight. Drawbacks

Propellant	Ideal I_{sp} (s)	Comments
LH_2	770 - 940	Low density cryogen
LHe	540 - 660	Medium performance cryogen
NH_3	260 - 320	Low performance storable

Table A.1 - Propellant Choices for Direct Thermal Propulsion (2000-3000 K temperature range)

of using LH_2 are high storage tank volume (LH_2 density = 70 kg/m^3) and thermal management problems associated with keeping the liquid hydrogen at low temperature for long times. Despite storage difficulties, the promise of high specific impulse (600-1000 s) makes hydrogen the only competitive propellant. Table A.1 above illustrates this.

The choice of nuclear fuel is less simple. Of the three main classes of uranium compounds, only the carbides and oxides are discussed here; the nitrides exhibit 'chemical instability' above 2000 K. There is limited performance data base on any fuel type above this temperature. High-temperature fuel performance is an area requiring more study.

Lundberg and Hobbins [Ref. 17] detail the high-temperature fuel issues to include, in order of importance:

(1) Material evaporation,
(2) melting temperature,
(3) uranium density,
(4) effects on reactor neutronics,
(5) high temperature chemical stability,
(6) fabrication difficulty,
(7) fission product release,
(8) fuel swelling,
(9) high-temperature creep,

(10) thermal shock resistance, and
(11) mass density.

Oxide fuel forms are extremely volatile above 2000 K and must be enclosed to prevent evaporation. Tungsten cladding or cermet (a metal matrix of UO_2 and W) has been suggested as possible containments. Evaporation is not the only issue; hydrogen attacks UO_2, liberating liquid uranium at the temperatures under consideration. This material is highly corrosive. All tungsten isotopes but W-184 have fairly high thermal neutron capture cross-sections; this makes the use of cermet more likely in fast-spectrum reactors. One form of cermet, W-60 v/o UO_2, has a melting point of *3075 K* [Ref. 17].

The carbides have similar evaporation problems but their use is complicated by the lack of a compatible container material. ZrC, HfC, and TaC have all been suggested; while ZrC has the lowest melting temperature of these, it exhibits the smallest neutron capture cross-section and is least likely to interfere with reactor neutron transport. One TaC-C eutectic has a melting point of almost 4300 K.

Carbide lifetimes are measured in hours; NERVA designers predicted operating lifetimes for UC-ZrC mixtures of ten hours or less at 3000 K (Ref. 18). This is acceptable when the total operating time for the system is very short; impulsive burns in low earth orbit last less than fifteen minutes. A Mars round-trip might require up to an hour of burn time at the high thrust levels being considered for the Space Exploration Initiative. The bimodal satellite hybrid reactor will be operating in a propulsion mode for hundreds of hours, limited only by available propellant.

No known fuel forms allow high-temperature operation (3000 K) for such long periods. In addition, it is unlikely that static power conversion systems (thermionics or thermoelectrics) can be modified to operate under these extreme conditions (temperatures above 2300K).

Appendix B Mission-Independent Design Issues

This section discusses the storage of cryogenic hydrogen within current launch system payload shrouds, and the dissipation of decay heat from a bimodal reactor system. While there are a number of other design issues that could be addressed in this appendix, these problems were fairly easy to analyze.

At 70 kg/m^3, liquid hydrogen is fourteen times lighter than water. A simple analysis was performed to determine if current payload shrouds would be capable of carrying both a heavier payload (enabled by the bimodal system) and the cryogenic hydrogen propellant needed to reach operational orbit. Table B.1 illustrates the various boosters' volumetric constraints and the total payload/propellant volume needed. The payload masses used in the table were taken from figures in section 3.1 and assume a bimodal system performance of 770 s I_{sp} and a thrust-to-weight ratio of 10^{-4}. Volumes were estimated using the algorithm *Volume = .01(Mass)*, taken from Ref. 7. The propellant tank was arbitrarily assumed to take up 15% of the propellant volume. Propellant mass is the weight of the orbital transfer propellant only.

Booster	Approximate Shroud Vol. (m^3)	Injected Weight (kg)	Propellant Mass (kg)	Total Volume Required (m^3)
Delta II 7920	43	2250	2850	69
Atlas IIAS	87	3800	4760	116
Titan III	113	6500	8000	196
Titan III Dual Payload Shroud	146	6500	8000	196
Titan IV 56'	226	10000	12500	305
Titan IV 66'	276	10000	12500	305
Titan IV 76'	326	10000	12500	305
Titan IV 86'	376	10000	12500	305

Table B.1 Volumetric capabilities of various boosters
(Assumptions are stated in the preceding paragraph.)

As can be seen, none of the current shrouds appear to be capable of containing bimodal satellite systems with large propellant tanks, save for only the outsize Titan IV shrouds (shaded). Performing bimodal missions will necessitate a redesign of the payload envelopes to meet the higher volume requirements.

The second issue is dissipation of waste heat produced following reactor shutdown. This heat is generated by the decay of fission products and can continue for some time after the reactor is turned off. Thus, the thermal management subsystem must be capable of removing this heat from the core and radiating it to space; if the decay power is too high for the thermal management to handle, then additional propellant would have to be expended to remove this waste heat until the decay power decreased enough to allow the radiator to handle it alone. The rate of energy generation from fission product decay can be described by the equation [Ref. 19]:

$$P(t) = 5.8 \times 10^{-3} P_o \left[t^{-0.2} - (t + t_o)^{-0.2}\right]$$

The decay power P(t), in watts, is determined by the total time after shutdown in days (t) and the total operating time (t_o) at a steady thermal power level P_o.

For a given thermal power level, the thermal management can be constructed to accomodate the decay power produced. As an example, consider a bimodal system with an electrical power output of 10 kW_e and a conversion efficiency of 10%. The reactor thermal output would therefore be 100 kW_t and the radiator would be sized to dissipate this amount of heat. Since, while operating in thermal propulsive mode, the bimodal system will be expelling its heat with the propellant, the thermal management system will play no role. Once the propulsion has been turned off, though, the reactor will continue to generate heat and the heat generation rate must be less than 100 kW_t to allow the radiator to remove it from the core. Given the assumed decay power, the equation above can be solved for Po, yielding a result of 1.78 MW_t. This translates into a thrust level of approximately 470 N at a specific impulse of 770 s.

Since the specification on thrust--detailed in section 5.0--is only 50 N, it is clear that the radiator is oversized for the thrust level and is driven by the electrical power requirement.

Appendix C Determination of LEO-GEO Delta-V Requirements

Most of the orbital mechanics work was performed on an equation-of-motion integrator developed by one of the authors for propulsion systems without impulsive transfer capability; the thrust-to-weight levels examined are between those exhibited by electric thrusters (~10^{-5}) and chemical propulsion (~1-10). The typical assumption for high-thrust chemical systems is to regard the burn as having taken place over essentially zero time. Other limiting assumptions can be used in the case of extremely low thrust. This integrator, EQMO, requires input in the form of initial vehicle mass, initial orbit specifications, propulsion system specific impulse and thrust level, and the desired final orbit altitude. EQMO does not perform plane changes or allow variable thrust profiles throughout the mission, although it does provide for impulsive maneuvers for verification of the program's accuracy. The thrust profile is assumed to be circumferential throughout the burn; the thrust vector is always perpendicular to the gravity vector. Program output consists of final orbit specifications, circularization delta-V requirements at the final orbit, transfer time, and burnout mass delivered. The program accuracy was checked against results [Ref. 20] for a low thrust spiral out maneuver and at high thrust-to-weight levels against impulsive LEO-GEO transfer delta-V requirements.

The integrator solves the equation of motion of the vehicle in order to determine its radial position--and thus altitude. Once an initial Schuler frequency (ω_S = initial velocity / initial altitude) is determined from the orbital parameters, a change in frequency can be calculated from the circumferential equation of motion. For a stated time step, a new frequency is found; this is used in the radial equation of motion to solve for the radial acceleration. An explicit Euler scheme was used to determine the radial velocity and then the radial position. This new radial position is introduced into the original calculation of the Schuler frequency and the iterative process repeated. Intermediate orbital elements are provided during the run. When the destination radial position is reached, the program outputs the material mentioned in the previous paragraph and ends the run.

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